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DESIGN CONSIDERATIONS OF FUTURE

SHORT HAUL TRANSPORT AIRCRAFT

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(Submission for Ph.D. by published work)

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SUMMARY

The rapid escalation of air transport in the past two decades has resulted in difficulties associated with airport congestion and ground trip delays. These have been particularly acute for short haul operations. Suggestions have been made that they might be alleviated by the use of smaller airfields located nearer to population centres than existing large airports which operate aircraft specifically designed for the purpose. This paper reports a systematic investigation of the requirements, design parameters and performance characteristics of short haul transport aircraft. Designs intended for operation in the VTOL, STOL, RTOL or CTOL modes have been considered and compared in an attempt to predict the more promising prospects for the future. A number of specific design studies were used as the basis for the comparison of the weight, economic and noise characteristics of the aircraft concepts which fall into the categories investigated.

The more significant conclusions of the study are:-

- (1) Genuine STOL operations from runways of the order of 2000 ft in length require a transport aircraft to employ some form of power augmented lift. Significant design and operational difficulties are implied by this. These are associated especially with noise, engine failure considerations and the relatively low approach and take off speeds. The augments wing and overwing blowing concepts are likely to prove to be the most suitable of the possible power lift concepts. However there is considerable doubt as to the need for this class of performance.
- (2) VTOL operations eliminate some of the low speed difficulties associated with STOL. The price of this is essentially economic. Although it is mechanically complex the tilt wing/rotor VTOL concept is potentially advantageous in terms of noise and fuel usage relative to the alternative fan lift aircraft.
- (3) An RTOL aircraft for 4000 ft runway operation is feasible and has advantages in that it gives the most scope for noise reduction. Development would be relatively straightforward in that only mechanical high lift devices would be required. The basic design could be developed to either a higher performance CTOL type, or by the addition of lift fans to V/STOL aircraft.

A design study of an aircraft which meets the RTOL requirements is presented.

PREFACE

The investigation reported here covers work undertaken by the author over a period of five years commencing in 1970. Certain of the specific aspects of the work were contributed by MSc students as their thesis research under the supervision of the author and a member of staff, R.E. Ward, has also been involved under the general direction of the author. Reference is made to these contributions where appropriate.

Much of the work completed has been of relevance to studies being undertaken both in Europe and North America and therefore the opportunity was taken to publish the results as soon as a particular section had been completed. This has been done within the Cranfield Report Aero series and the relevant documents are included as Part 2 of this submission. They are:-

- | | |
|--------------|---|
| Report No.12 | Aircraft Design Studies - STOL Airliner
(June 1972)* |
| No.18 | Performance characteristics of short haul
transport aircraft intended to operate
from reduced length runways (April 1973) |
| No.24 | The weight economic and noise penalties of
short haul transport aircraft resulting
from reduction of balanced field length.
(Jan.1974) |
| No.25 | Subsonic jet transport noise, the relative
importance of various parameters (July 1974) |

Several of the aircraft configurations compared were used as student design study examples. Those which the author was responsible for preparing and supervising are described in detail in Report No.12 and Appendices D and E. In common with Reports Nos.18, 24 and 25 these present only the results and conclusions of the work, the extensive intermediate calculations being omitted for conciseness.

Some of the background work has been used as lecture material for the Aircraft Design course and Appendices A, B and C are largely based on lecture notes which are currently in use.

In order to give continuity to this submission it has been necessary to extract some of the text and figures from the Appendices and Part 2. For the same reason all the references used in the study have been gathered together irrespective of whether they have already been quoted elsewhere in the subsidiary part of this report. However in spite of these provisions the main text of the submission must be regarded as a summary of the work undertaken and it is essential to read the published reports of Part 2 in conjunction with the relevant sections of Part 1.

*Written jointly with R.E. Ward who was responsible for preparing some of the aerodynamic data, see Section 8.5

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FIGURES

Part 2 Published Reports.

Cranfield Rep. Aero. No.12	Aircraft Design Studies - STOL Airliner.
Cranfield Rep. Aero. No.18	Performance Characteristics of Short Haul Transport Aircraft intended to operate from reduced length runways.
Cranfield Rep. Aero. No.24	The Weight, Economic, and Noise Penalties of Short Haul Transport Aircraft Resulting from Reduction of Balanced Field Length.
Cranfield Rep. Aero. No.25	Subsonic Jet Transport Noise, the Relative Importance of Various Parameters.

PART 1

SURVEY OF INVESTIGATION

1.0 INTRODUCTION

The era following World War II has seen substantial improvements in the safety, reliability and economy of civil transport aircraft. When these are coupled with the convenience offered by the speed of air travel the explanation for the vast increase in air traffic is obvious. The developments in air transport have not been achieved without some problems asserting themselves, but as is so often the case it is difficult to distinguish between cause and effect. The technological improvements in the vehicle have been obtained in part by increases in both size and weight. This has been possible in spite of the well established effect of the square-cube law which militates against larger aeroplanes. There has been a continuing tendency to increase wing loading. The consequent aggravation of the low speed performance has only been partially offset by improvements in high lift devices. It is of interest to note that during the past three decades the weight of the largest operational transport aircraft has increased at the rate of approximately 200,000 lb per decade at the expense of a corresponding increase in wing loading of 30 lb/sq ft per decade. An inevitable result of these trends has been the need to provide the longer runways necessary to handle aircraft having higher take off and landing speeds. It is not surprising to find that the tendency here has been in line with aircraft developments at the rate of some 2000 ft increase per decade. It is questionable whether this trend can continue if only due to lack of available ground for significant further extension. Already the necessity for long runways has resulted in a tendency for major airports to be located at increasing distances from the centres of population which they are intended to serve. As a result of this ground transit times have become greater with a general worsening of traffic congestion and the extra time taken to handle large aircraft at extensive major airports has aggravated the situation. Further time delays have occurred because of air traffic congestion in terminal areas as a direct consequence of the much higher frequency of operations. Thus the air traveller has been forced to contend with the frustration of long terminal delays and one of the major advantages of air travel, namely speed, has become seriously eroded relative to ground transport. This is especially true on shorter routes. At the same time the expansion of air transport operations in the vicinity of large centres of population has manifested itself as a major environmental problem. This is primarily due to the noise nuisance which has derived from the greater number of flights by aircraft of ever increasing size and power.

Summarising therefore it may be stated that amongst the main problems faced by air transport at the present time are:-

- A) The need to reduce the total 'ground' time to an acceptable level, especially in the case of short haul operations.
- B) The desirability of easing air traffic congestion at peak periods.
- C) The reduction of community interference.

Of course it goes without saying that any improvements made in these directions must not detract from safety or reliability, or for that matter economy in the present environment of high fuel costs. However in this latter case it is not sufficient to consider the operating costs of the aircraft in isolation from those of the total system.

During the late 1960's it was suggested that one way of tackling these problems would be to develop a new class of short haul airliner having the capability to operate quietly from sites much nearer to city centres than existing international airports. The impetus for this concept came primarily from the United States where on occasions during peak periods the air traffic density has been such as to cause a breakdown in regular scheduled operations. Experiments were undertaken by American Airlines, reference 1, in which a number of existing aircraft with short runway performance were operated out of the centre of New York City. These lead to the preparation of a draft specification for a suitable new aircraft, and consideration being given to the provision of operating strips by such bodies as the Port of New York authority. Feasibility studies for both the vehicles and overall systems were initiated by various organisations in North America, references 2, 3, 4 and 5. Naturally to some extent this work was biased by the particular interests involved.

During the same period studies were undertaken in the United Kingdom along somewhat similar lines, with the addition of an emphasis on the possibility of vertical rather than short take off and landing performance, reference 6. This was considered to offer appreciable advantages in reducing size of ground facilities with the implied greater choice of site location, and the prospect of a lower community noise nuisance. The concept was a logical step from the earlier pioneering work on jet lift initiated by Rolls-Royce with the 'Flying Beadstead' and demonstrated in application to a practical aircraft by the Short SC1.

In the realm of military operations there has always been a need for tactical transport aircraft capable of operating from relatively poor quality fields close to the combat zone. Whilst the design problems associated with this class of aircraft differ in some respects from those of civil transports there are also some important similarities. Amongst these may be mentioned the desirability of keeping the required runway length to a minimum and the need to keep noise as low as possible to avoid alerting the enemy. It is of interest that some of the United States work mentioned above has been diverted in the direction of the tactical military transport and eventually could form a basis for civil developments.

At the beginning of 1970 a very confused picture of the future of short haul transport was presented. In spite of the work being undertaken at that time it was not clear whether any new class of aircraft was likely to be an economically feasible possibility or indeed whether such an aircraft was required. Nor was it apparent whether vertical or short take off and landing concepts provided potentially the best solution, or for that matter which of the various possible ways of achieving these characteristics was preferable. In fact in many cases the overall characteristics of the various classes of V/STOL aircraft were unknown. The present work was undertaken in an attempt to resolve some of these issues. It has been integrated with the academic programme in the Aircraft Design Division of the College of Aeronautics. In this way certain aspects of the overall study could be investigated by MSc students who would thus obtain some benefits from the whole exercise. In particular it has been possible to use some of the individual aircraft design studies as bases for group project work.

2.0 PURPOSE OF INVESTIGATION

The fundamental purpose of the study has been to establish the likely future trend in short haul transport. In order to achieve this it was necessary to undertake a number of more specific tasks:-

- A) The establishment of the basic performance characteristics of aircraft intended to operate from reduced length runways.
- B) The influence of these characteristics on the overall and detail design of the aircraft, and especially their effect in causing departures from the design and operational trends associated with conventional designs.
- C) The penalties associated with reduced runway performance
- D) A comparison between the various forms of V/STOL aircraft and more conventional configurations in order to deduce the likely trends for the future.

A secondary but nevertheless important reason for the investigation was the integration of the relevant academic and research activities of the Aircraft Design Division.

3.0 METHOD OF APPROACH

The means used to tackle the investigation was essentially that adopted in all aircraft initial design work. Inevitably the process was an iterative one with the first steps determined on the basis of past experience. The stages in the work were:-

- A) A statement of a specification as a basis of comparison. To a large extent reliance was placed on the work in both the United States and the United Kingdom for this although in one particular respect, that of noise requirements, it was considered that the basis of comparison warranted further consideration.
- B) A parametric investigation to determine the interaction of the most important design features.
- C) The detailed investigation of a series of "referee" designs to check the assumptions of the parametric study and correct them where necessary. These designs also formed the basis of the overall comparison and therefore needed to be sufficiently detailed to justify reliance being placed on the results. A great part of the total effort expended on the study occurred in this aspect of the work, both in the initial synthesis and in the detail design needed to confirm it. A considerable contribution was made by MSc students under the supervision of the author.
- D) The comparison of the V/STOL and conventional designs from which future trends were deduced.

In any work of this nature it is inevitable that many assumptions and simplifications are made. Experience plays a large part in this process and there is a tendency to extrapolate trends from known aircraft designed to meet similar, but not identical, requirements. The dangers inherent in this when dealing with novel configurations are appreciated and therefore a critical approach has been adopted throughout the study.

4.0 BACKGROUND STUDIES

Several design studies which were undertaken prior to the start of the present investigation afforded some indication of the problems likely to be encountered. Three of these are worthy of specific mention since the lessons learnt from them had a substantial influence on the subsequent work.

4.1 1959 Short Take Off and Landing Freighter

The earliest work dates from 1959 when a project study of a large STOL freight aircraft was undertaken. This work has been fully reported in reference 7. A general impression of the aircraft can be gained from figure 1. The aircraft was intended for use either as a civil transport or in a military role. The pressurised cargo compartment was of sufficient volume to allow low density loads of up to 34 tons to be carried and at the same time the floor strength catered for large concentrated loads such as heavy earth moving equipment. The cargo hold could be loaded either through a nose door or through a rear ramp door which also served for the air dropping role. The fuselage mounted bogie undercarriage used large low pressure tyres and the resulting runway requirement was an LCN of 25 at the maximum weight of 200,000 lbs.

STOL performance was obtained by using boundary layer control in conjunction with the moderate wing loading of 90 lb/sq ft. The boundary layer control took the form of blowing over the slotted flaps and drooped ailerons. The air required for this was obtained from a pair of auxiliary power units housed in the undercarriage blisters supplemented during landing by compressor tappings off the four Rolls-Royce Tyne powerplants. The approach speeds were predicted to be 95 knots at the maximum landing weight of 190,000 lbs and 80 knots at a weight of 130,000 lbs.

Although the aircraft had a predicted maximum level speed of 368 knots at 26,000 ft altitude the best cruising condition was 310 knots true speed up to 30,000 ft altitude. The range available varied from 750 n miles at maximum payload to 5700 n miles with maximum fuel and 9 tons payload. Both take off and landing could be accomplished in 2000 ft but the factored field length was nearer 3000 ft. The aircraft was equipped with large stabilising and control surfaces to meet the requirements of a large centre of gravity range and the air dropping role. Because of this it was considered that there would be no severe low speed control problems especially as the approach speeds predicted were not unduly low.

4.2 1961 Vertical Take Off and Landing Freighter

This study was a direct derivative of the previous design discussed above. Reference 7 also contains a full report of the investigation. The major differences between the two designs are readily apparent by comparison of the general arrangement drawings, figure 2 being that for the VTOL version. The aircraft was conceived as a hybrid between a conventional design and one with VTOL capability. For this reason the vertical lift was provided by a total of no less than 44 fan engines which were housed in two detachable wing mounted pods. Each lift engine had a bypass ratio of 3.5 and a thrust rating of 8000 lbs giving a total nominal vertical thrust of 352,000 lbs. The corresponding maximum weight of 250,000 lbs represented an increase of 50,000 lbs over the previous STOL version. The lift pod assemblies accounted for 44,000 lbs of this and the remainder was additional fuel. The presence of the pods on the wing reduced each flap span by some 18 ft and in order to achieve acceptable conventional low speed performance without boundary layer control the flaps were changed from single slotted to Fowler. The resulting approach speed of 112 knots at 190,000 lbs weight implied the need for a 5000 ft balanced field length.

With the VTOL pods fitted the design performance criterion was the carriage of 35 tons payload over 400 n miles range when take off and landing vertically at 5000 ft altitude in a temperature 15°C above I.S.A. Sufficient fuel was carried to enable two complete VTOL cycles to be undertaken to cover the case of a landing being aborted immediately prior to touchdown. In the VTOL configuration the maximum level speed was calculated to be 322 knots at 17,000 ft altitude, the normal cruising speed being 225 knots equivalent airspeed between 15,000 ft and 20,000 ft altitude.

For the hot and high take off conditions water injection was provided to boost the lift engine thrust to 1.25 times the weight. Of this 25% excess some 10% was used solely for vertical acceleration and the remaining 15% for control. The control in vertical and transition flight was obtained directly from the lift engines. The engines in each pod were divided into two groups, one forward and one aft of the wing structure. Pitch control was obtained by symmetric fore and aft differential thrust of the groups whilst roll control required asymmetric differential thrust between the two pods. For yaw control deflector buckets on the engine nozzles were moved about a lateral axis to give fore and aft differential forces between the two pods. No auxiliary nozzles were necessary.

During transition from vertical to forward flight the pitch altitude was automatically stabilised at a safe angle below the stalling condition with the flaps deployed. The propulsion engines increased the forward speed and an automatic progressive reduction of lift thrust was used to reduce the momentum drag as much as possible. At the same time control was shifted to the conventional trailing edge surfaces so that total control forces available were adequate. Transition was complete at about 130 knots and at this stage the lift engines were shut down and the intake and exhaust doors were closed. The whole process of transition required just under one minute. Provision was made in the performance, control and structural calculations for the failure of any two lift engines in either pod.

A number of important conclusions resulted from this investigation:-

- A) The noise level of the relatively low bypass lift engines was exceedingly high. The predicted level of over 160 dB on the 500 ft sideline was intolerable on any basis. The implication of this was the need for a much higher bypass ratio than the 3.5 used for the lift engines and the addition of acoustic treatment techniques not available at the time of the study.
- B) The basic layout of the aircraft dictated that the lift engine pods had to be located well outboard on the wing span, between the flaps and ailerons. One serious consequence of this was the severe fatigue damage at the wing root resulting from the magnitude of the ground-air-ground cycle, in spite of the fact that the in-flight relief moment due to the pod compensated for the higher all up weight of the aircraft. In fact the wing safe life in the VTOL role at design weight was predicted to be as low as 1700 hours.
- C) A further consequence of the location and size of the lift pod was manifested in wing aeroelastic problems. The wing torsion frequency was dominated by the immense inertia of the pod and the usual practice of increasing wing stiffness to raise flutter speed proved to be of little avail. Subsequent investigations at Loughborough University showed that the solution to this difficulty lay in the use of pod tail surfaces to provide aerodynamic damping moments in those cases where the pod could not be moved inboard.

4.3 1967 Tilt Wing Executive Aircraft

The third of the background projects was in a completely different category to the two large freighters. It was a comparatively small VTOL aircraft intended primarily for executive and third level feeder line operations. The overall concept of the design is shown in figure 3 and the work has been fully reported in reference 8. The layout was based on the use of a twin rotor tilt wing arrangement. Although

intended primarily to fulfil the executive role the 19 ft long cabin was designed to be capable of accommodating up to 18 passengers for third level airline operations. In this high density role there were six rows of three seats with a dividing aisle having 6 ft clear headroom. The nature of the tilt-wing concept dictated a high wing for reasons of ground clearance and this resulted in a 7 ft external diameter for the pressurised fuselage.

As originally designed the aircraft was powered by two 1400 HP Rolls Royce Gnome H 1400 shaft turbines. Each of these drove a 16 ft diameter propeller through reduction gearing mounted at the rear of the engine. A power offtake from the reduction gearbox was used to interconnect the two propellers, the whole arrangement being very similar to that employed for propeller interconnection of the Breguet 941 aircraft. A central gearbox on the cross shafting was provided for accessories and a drive for a small horizontal pair of contrarotating tail rotors. As the design study progressed it became apparent that whilst the propeller interconnection was valuable in alleviating control problems after failure of an engine in vertical or transition flight, the remaining engine was of insufficient power to maintain altitude when the forward speed was zero. It was therefore considered to be essential to revise the powerplant arrangement and replace each of the Gnome engines by a pair of interconnected smaller units, such as the Garrett TPE 331 shaft engine. This would confer four engine reliability without changing the configuration, as there would still be only two propellers, and also enable the aircraft to operate safely in the event of single engine failure.

The design vertical take off weight was 13,000 lbs of which some 4135 lbs was predicted to be available as disposable load. A maximum design weight of 15,000 lbs was used for short take off and landing operations. The disc loading in the vertical take off case was 32 lb/sq ft whilst with the wing area of 260 sq ft the wing loading was 50 lb/sq ft. The wing incidence could be varied from +2 degrees to +102 degrees relative to the body datum. Kruger flaps were placed along the whole of the leading edge and 35% chord double slotted flaps positioned along the trailing edge inboard of the powerplant nacelles. The low mounted tailplane was of the all moving type capable of being operated over the range of 15 degrees down to 45 degrees up.

During cruising flight the aircraft was controlled in a conventional way using the tailplane, rudder and ailerons. Whilst the aircraft was in vertical and transition flight modes the control forces were produced by a combination of rotor and aerodynamic surface effects. Roll control was achieved simply by differential use of the collective pitch change in the two propellers. Simultaneous collective pitch change used in conjunction with throttles was employed to control vertical motion. Longitudinally use was made of the contrarotating tail rotor in conjunction with overall collective pitch variation. When the propellers were in the horizontal plane the ailerons were aligned vertically

beneath them so that they could be used for yaw control. As an alternative to this system consideration was given to rotation of the tail rotor about a fore and aft axis to give a lateral thrust component. This concept was not found to be essential. Tilting of the wing and hence the overall thrust vector gave fore and aft control.

Transition from hovering to forward flight was accomplished by a gradual tilt-forward of the wing to increase the forward speed. As the wing started to rotate towards the horizontal position the slipstream and forward speed effects enabled a vertical component of aerodynamic force to be developed to offset the loss of vertical powered lift. This process was assisted by the programmed deployment of the leading and trailing edge flaps. The transition was effectively complete at 60 knots forward speed with the wing angle at approximately 12 degrees. At this speed there was still a very large beneficial slipstream effect on the wing which was some 16 knots below the estimated zero power stalling speed. During the wing tilt process the role of the ailerons was automatically transferred progressively from yaw to conventional roll control.

The estimated vertical thrust during vertical take off was 14000 lbs, the margin over the take off weight being allowed for acceleration and control. The power installed to provide this vertical thrust was some 40% greater than would normally be expected for a conventional aircraft of comparable weight and role. Hence the forward speed performance was better although the effect of the extra power available was somewhat offset by a lower cruise propeller efficiency due to the compromise with take off performance. The usual cruise condition was predicted to be some 285 knots true airspeed at 20,000 ft altitude. The maximum normal cruising speed at low altitude and 9000 lbs weight was estimated to be 290 knots. In the basic VTOL condition a 2000 lb payload could be lifted over 500 n miles still air range, no reserve allowance. With short rather than vertical take off this was increased to 1100 n miles.

A number of conclusions were made as a result of this investigation, the more important of which can be summarised as:-

- A) A four engine arrangement is essential for civil operation of this class of aircraft. In this configuration the safety of the aircraft should be at least as good as that of a twin engine helicopter.
- B) The major development problem associated with the aircraft would be the transition aerodynamic performance.
- C) The use of a tail rotor with mechanical drive was simple in concept but did introduce a number of layout and reliability problems. Because of this a study of alternatives such as cyclic pitch variation of the main propellers or the use of reaction controls, was indicated.

5.0 SPECIFICATION OF SHORT HAUL TRANSPORT

In the post war era the major improvements in the design of transport aircraft have been in the direction of higher speed and greater payload, with which has been associated reduction of real operating costs. Passengers have come to expect a high standard of comfort, partly as a legacy of the first class image of early air transport and partly because of the smooth flight of high altitude turbine aircraft.

In a lecture delivered a few years ago, Steiner, reference 9, suggested that in future transport aircraft would have to be designed with a different emphasis than that of the past. He named the prime design considerations as noise, comfort and economics in that order. This statement was made before the rapid increase of fuel costs which have occurred recently but there is no fundamental reason why the emphasis should be changed. Whilst noise levels have been of concern for some while early improvements were not allowed to introduce major performance penalties. Indeed the use of such techniques as higher bypass ratio has resulted in noise reduction associated with lower fuel consumption. Passenger comfort has also been, to some extent, a by-product of improved performance. Safety considerations should perhaps be added to the list given by Steiner since the increased frequency of operations and larger capacity aircraft must be accompanied by a steady reduction in accident rate if public confidence in air transport is to be maintained.

A set of requirements must be quantified if a meaningful comparison between different forms of short haul transport is to be made. Bearing in mind the current trends it is important to stipulate target noise and comfort levels as well as the more usual speed, range and payload performance. The basis of comparison used in this investigation was as follows:-

A. Range.

In general short haul may be regarded as covering flights up to around 1000 n miles in length, but operating experience shows that the bulk of them are less than half this distance. Therefore the basic range requirement with full payload was set at 600 n miles. This can also cover the case of two 250 n mile stage lengths without intermediate refuelling. The reserve fuel allowance depends to some extent upon the type of aircraft but typically the additional fuel necessary to cater for this enabled a zero reserve still air range of about 1000 n miles to be achieved with full payload.

B. Payload.

Experience has shown that short haul transport designs for scheduled operations typically have a 30% increase of passenger capacity in each new generation. There are, of course, exceptions to this especially with the introduction of wide body aircraft. In the present investigation it was considered desirable to be able to make direct comparisons with aircraft of current size which have payloads of the order of 100 to 120 passengers. Therefore in the comparative work a standard of 108 passengers was taken as the full payload.

C. Cruise speed.

To some extent the cruise speed is influenced by the type of aircraft and this was compensated for in the comparative studies. However a cruise speed of $M = 0.8$ was selected as a datum for initial design work.

D. Noise.

There are various ways of establishing a criterion for noise comparisons. This subject is discussed more fully in Appendix A and it is sufficient to comment here that the main choice lies between the use of existing certification requirements such as FAR Part 36 or a noise footprint area. Since it is considered that the ultimate noise target should be to minimise the 80 PNdB noise footprint area, and preferably limit it to within the airport boundaries, this was selected as the basis for noise comparison.

E. Passenger comfort.

Two aspects of the passenger comfort must be considered in relation to the overall design and performance of the aircraft. One is the question of tolerable peak accelerations which occur at specific times during the flight, such as take off and landing. The other is the ride characteristics of the aircraft in cruise. It was considered that any new short haul transport should operate in such a way that the passengers would not be subjected to more severe conditions than with the current generation of comparable types. The criteria established to ensure this are discussed in section 7.1.

F. Safety.

Novel configurations can introduce methods of operation and other considerations which require modification to established requirements. The aircraft studied in this investigation were designed to meet British Civil Airworthiness Requirements Section D, (Conventional aircraft), Section G (Helicopters), and where appropriate, the draft Section P (Powered lift aircraft). In one or two instances none of these was found to be adequate and it then became necessary to introduce appropriate conditions.

6.0 INITIAL DESIGN DATA

A major part of the investigation was concerned with the collection of initial design data and its correlation into a form suitable for initial design work.

6.1 Aerodynamic Data

The aerodynamic data used in the studies was derived from two main sources:-

- A) Existing documents relevant to conventional aircraft and conventional features of some of the more unusual configurations. The main reference in this group was the ESDU Data Sheets in the Aerodynamics series.
- B) The series of NASA and associated reports of wind tunnel tests and design studies of various classes of STOL aircraft, references 14 to 29. In general these reports presented information for particular aircraft layouts and the results were not directly applicable to the general study in hand. The task of converting the information into a more useful form was commenced by the author but much of it was the work of R.E. Ward, references 30, 31 and A.S. MacKichan, reference 32.

6.2 Weight Prediction Data

The author has been systematically collecting and analysing aircraft weight data for more than two decades. As long ago as 1957 an attempt was made to derive both empirical and theoretical formulae for initial design work, reference 33. However since that time there has been a rapid development in aeronautical technology and it was necessary to update the techniques. A completely new analysis was undertaken for the present work and the resulting formulae are stated in Appendix B. These have been based on the correlation of empirical data and the adaptation of theoretical analysis using that data. Some information on the weight penalties associated with quiet augmenter wing systems was obtained from reference 29.

6.3 Noise Prediction

In many respects the prediction of aircraft noise characteristics introduces difficulties which are similar to those encountered in weight prediction. An adequate theoretical basis is difficult to establish because of the complex interaction of many parameters. If a satisfactory solution to this problem is eventually achieved it will require a knowledge of parameters which are not completely established in the initial design phase. Therefore it is necessary to rely upon experimental data and to attempt to use a theoretical framework to interpret it. A number of investigators have done this for various aspects of powerplant, rotor, and more recently, airframe noise, references 38 - 52. For the purposes of the present study the author reviewed the published work and adapted it to suit the particular

requirements of the investigation. The airframe self noise aspect was one which assumed considerable importance for some of the configurations studied and at the time when this part of the investigation was being undertaken no applicable prediction techniques were available. The author therefore produced his own techniques. The noise of the powered lift systems also presented some problems and these were overcome by analysing, correlating and reconciling the results from various test programmes, most of which were undertaken under the auspices of NASA, references 53 to 55.

The noise prediction part of the study and the relevant formulae and methods used are covered in Appendix C.

6.4 Economic Prediction

The basis of the Direct Operating Cost Analysis was an existing method developed by British Airways for short haul operations, reference 56. It was necessary to adapt the technique to cover the cases of unusual configurations especially those using various forms of powered lift. The assumptions made are stated in Part 2, Cranfield Rep. Aero. No.24, Appendix B, reference 71 was also used.

7.0 PERFORMANCE CHARACTERISTICS

Before commencing the detail investigation of the specific short haul transport concepts it was necessary to undertake a general analysis to establish the main parameters involved and their relative importance. In particular a clarification of the effect of reduced runway length on aircraft geometry and performance was required. This phase of the study has been reported in Cranfield Report Aero No.18 which is to be found in Part 2 of this document. Four specific characteristics were considered.

7.1 Passenger Comfort

The cruise ride characteristics of existing transport aircraft were analysed in the context of the sharp edge gust formula and related to known passenger preferences. From this it was possible to produce a simple formula relating the cruise speed, wing loading, aspect ratio and sweep back of an aircraft having acceptable ride characteristics. Aspect ratio and sweep back directly affect the lift curve slope of the wing but relative to the other two parameters their effect was of less importance. Thus basically the formula enabled a minimum acceptable cruise wing loading to be specified for a given cruise speed, Rep. 18, figure 1.

A study of the tolerance of human beings to fore and aft acceleration using results from ground transport research was related to aircraft take off and landing behaviour, Rep. 18, figure 2. The main restriction on aircraft performance occurs when the acceleration is tending to throw the passenger from his seat. This coincides with the landing conditions for conventional forward facing seats. Analysis of the stated design landing distances of current aircraft suggested that the values of stopping deceleration normally employed at present represent the tolerable limit in terms of passenger comfort. The actual design deceleration available is of the order of 0.33g although a value of between 0.21g and 0.23g is normally used, Rep.18, figure 6. The present study was therefore based on a maximum available deceleration of approximately 0.33g. In emergency conditions it was considered that 0.5g deceleration would be both acceptable and technically feasible.

Take off acceleration with the passenger in a forward facing seat is less restrictive and evidence suggests that 0.5g or even 0.6g is tolerable.

7.2 Approach Speed and Descent Angle

In the context of reduced length runway operation it is important that the approach speed should be as high as possible, primarily to reduce the effects of cross winds and air turbulence. Given a runway length the usable approach speed is determined by three main considerations, the glide slope angle, the flare conditions and the stopping deceleration.

The investigation considered the influence of glide slope angle and determined that there was little to be gained for descent angles in excess of about 7° , except possibly for very short runway operations. In any case there is an overriding limit on the descent velocity during approach. A commonly used figure of 1000 ft/min maximum descent rate implies that the glide slope angle must be less than 7° for approach speeds in excess of about 80 knots, Rep.18, Figures 3 and 5.

As far as flare conditions are concerned it was deduced that the normal acceleration in the manoeuvre should not exceed 0.25g for reasons of passenger tolerance. Consideration was given to the use of an incomplete flare but whether this is possible in practice is open to serious question.

The stopping deceleration limitation is discussed above..

Although the approach speed follows from these considerations the value of approach lift coefficient relative to the stall condition demands some comment. The accepted factor of 1.3 for the approach speed above the stall speed which is used for conventional aircraft may be considered as covering actual or indicated excursions away from the datum speed as well as giving an incidence margin and an allowance for rotation and horizontal gusts. When a low approach speed is associated with relatively high lift coefficients the use of the 1.3 factor implies a larger incidence margin but, of course, a lower absolute forward speed margin. Arguments have been made in favour of reducing the factor for STOL aircraft but the justification for this is not obvious and must be associated with the characteristics of an individual aircraft. For the sake of generality the factor of 1.3 was retained although a decrease or even an increase might be in order.

Applying the deduced limitations in conjunction with the landing wing loading derived from cruise passenger comfort consideration it was possible to estimate the approach speed and approach lift coefficient appropriate to operations from runways of various lengths. This included the influence of wing geometry and cruise speed. The analysis covered factored runway lengths in the range of 1500 ft to 5000 ft, Rep.18, figure 7 to 10. The corresponding approach lift coefficients were predicted to be in the range of about 5.5 down to just over unity depending upon wing geometry and cruise speed. The approach speed for a 2000 ft runway cannot exceed about 80 knots.

7.3 Take Off Requirements

The take off performance is more difficult to generalise than landing performance due to the added effects of take off thrust/weight ratio and engine failure possibilities. The latter was found to be especially important for powered lift configurations where both lift and propulsive thrust fall as the consequence of loss of power. In these cases the speed corresponding to the condition of lift with one engine failed being equal to the weight provided the basic criterion for the take off performance and normally overrode control considerations.

The problem of carrying out a general analysis of the take off conditions was tackled by separating the ground roll phase from that of rotation and climb out to clearance height. The wing loading at take off was determined in relation to the minimum necessary for cruise comfort by assuming that the normal fuel used in a flight would be equivalent to 11.5% of the take off weight. Variations of basic thrust during take off was estimated as a function of bypass ratio and the unstick speed at the end of the ground roll was related to the take off safety speed with one power unit failed. In order to do this it was necessary to make certain assumptions concerning the forward speed after rotation. The most significant of these was that longitudinal acceleration was zero during the period between rotation and reaching the screen height. Thus the initial climb out speed was equal to the take off safety speed which was taken as 1.2 times the stalling speed with one powerplant failed.

The ground roll distance to a given velocity was readily estimated as a function of bypass ratio and thrust/weight ratio using a typical drag characteristic, Rep.18, figure 11. Working back from a given runway length it was then possible to evaluate the corresponding values of the ratio of the unstick lift coefficient with one powerplant failed to the take off wing loading. This required the assumption of a given normal acceleration during rotation and implied the use of a certain braking deceleration in the event of an emergency stop following a powerplant failure at the rotation speed, Rep.18, Figure 12. Validity boundaries were then constructed for assumed emergency stopping deceleration of 0.5g, Rep.18, Figure 13. The presentation of the results was complex due to the large number of variables but the initial thrust/weight ratio required for safe take off from a given runway and for a particular cruise speed was plotted as a function of safe unstick lift coefficient, wing geometry and normal acceleration in rotation, Rep.18, Figure 14-17.

7.4 Low Speed Control

The low approach speeds predicted for shorter runway operations suggest that low speed control problems are likely to be severe. The effects of flight path disturbances due to gusting are approximately directly proportional to speed whilst the forces derived from conventional aerodynamic controls are proportional to the square of the speed. There is thus a general tendency towards a situation where aerodynamic controls become inadequate as forward speed is reduced, although the exact conditions naturally depend upon other design factors such as the size of control surfaces. It is also to be noted that shorter runway performance implies higher usable lift coefficients which can only be achieved by some means of power augmentation. For this reason the control problem associated with powerplant failure also becomes more severe at lower flight speeds but it is not possible to generalise the degree of this effect. Augmented control is implicit in VTOL configurations but the concept does eliminate those problems which arise from low, but finite, forward speeds.

These problems associated with low flight speed can be illustrated by reference to the operation of STOL aircraft from single runway sites where cross wind landing conditions can give rise to serious difficulties. A good idea of the severity of the problem may be obtained by reference to the equivalent sideslip angle resulting from the approach and cross wind speeds, Rep.18, figure 18. Conventional aircraft do not usually operate in conditions in excess of 12° equivalent sideslip angle, although it is considered that up to 20° could be accepted providing special provision is made in the design, such as a castoring main landing gear. The annual wind conditions at two typical locations in the United Kingdom were analysed and used to deduce a preferred runway direction in each case. One site chosen was a relatively sheltered one and the other was severely exposed. The number of hours per year when operations would have to be suspended because of sideslip angles in excess of the limits was calculated as a function of approach speed, Rep.18, figures 19 and 20. It was found that there would be no appreciable limitation on operations from the sheltered site providing the approach speed exceeded about 90 knots, but in the case of the exposed site provision for the 20° sideslip case was desirable. In this latter event approach speeds as low as 80 knots were found to be acceptable.

The provision of stability and control in these low speed flight conditions is a matter requiring particular consideration for each individual concept. In the overall study this was undertaken by Ward, References 30 and 67, for two particular STOL concepts. Some general comments on the results of these analyses are in order:-

- A) Longitudinally the main difficulty is associated with the provision of adequate trim and control on the landing approach. Typically for a 2000 ft STOL design the provision of a large adjustable incidence tailplane provided just about adequate trimming force with little or no margin for control. Therefore power augmentation, probably in the form of blowing over the tailplane/elevator combination is essential. The best design solution could be to limit the tailplane size to that determined by more conventional approach conditions and to provide augmentation as required for STOL conditions.
- B) Lateral instabilities in the spiral and dutch roll modes occur at lower speeds. These apparently arise from the change in the relative values of aerodynamic and inertial coupling. Modification of fin and dihedral characteristics seem to have little effect and hence autostabilisation is suggested.

- C) During cross wind approaches the directional control has to be sufficiently powerful to maintain the required heading and to provide adequate margin for manoeuvre and to cope with gusting. The magnitude of the latter can be of the order of 60% of the steady cross wind component and is superimposed upon it. The main design problem is that of providing the necessary rudder power without stalling the fin. A large conventional rudder is unlikely to be adequate, and would probably need to be associated with a blown fin to avoid stall. The solution could be the use of a blown fin and rudder, with very large rudder movement to offset the fin stabilising effect. Alternatively the use of canard rudders might be of value if the drag penalty is acceptable.
- D) Engine failure can give rise to large rolling moments in those cases where the power augmentation of the lift is not balanced across the span. The control problem is similar to that of the elevator in that blowing or some other form of augmentation is necessary. The use of blowing can enable large aileron angles to be used. Differential flap movement can produce powerful rolling effects but unfortunately is associated with substantial roll-yaw couplings which are likely to be unacceptable.

7.5 Conclusions

As well as the numerical results derived from this phase of the work it was possible to make a number of important general conclusions. These were:-

- A) Passenger comfort considerations have a significant effect upon the design and performance of aircraft intended to operate from short runways.
- B) Take off rather than landing determines the runway length, unless the runway requirement is very short.
- C) Mechanical high lift devices alone enable transport aircraft to be designed for operation from runways of as little as 4000 ft in length, but significant reduction below this introduces the need for some form of powered lift.
- D) Low speed control difficulties are serious and the configuration of the aircraft must be determined to minimise these.

8.0 REFERENCE DESIGN STUDIES

The performance characteristics discussed in the previous section were in the form of a parametric study which inevitably involved the making of a number of assumptions. The justification for these assumptions demanded a much more detailed study of the characteristics of typical aircraft. As well as this it was found that some important considerations in the performance, such as control problems at low speed, could not be generalised and were dependent upon the concept of an individual design. For these reasons it was essential to undertake a detailed design of a number of reference designs. This procedure had the additional merit of providing sufficient information to enable realistic comparisons to be made between the various design concepts available to meet the basic situation. The configurations selected were chosen on the simple criterion that adequate basic data had to be available to enable a realistic design to be prepared. They covered the range from conventional aircraft through reduced and short take off and landing to vertical take off and landing concepts. The investigation was primarily limited to designs using fan engines as experience has shown that propeller driven aircraft lack passenger appeal. An exception was made in the case of vertical take off. Brief descriptions of these designs were included for completeness in Cranfield Report Aero. No.24, which is included in Part 2 of this document. The leading particulars can be found in Table 1 of that report.

8.1 Powerplant

The performance and general design of any aircraft are both critically dependent upon the powerplant characteristics. Since the present study was forward looking it was essential to use data appropriate to the next generation of aircraft although at the same time it was essential for this to be adequate and realistic in order for the overall conclusions to be valid. These requirements presented some difficulties in the case of powerplant assumptions. At the time the study began there were proposals for new powerplant concepts for propulsion engines but insufficient data concerning them was to hand. It was therefore necessary to base the initial work on existing technology standards and that appropriate to the Rolls-Royce RB211 was selected. The bypass ratio of this engine is approximately five. Subsequently sufficient information of the Rolls-Royce-Snecma M45 development powerplant series became available. As far as the design studies were concerned the main differences between these proposals and the RB211 standard are as follows:-

- A) The bypass ratio is approximately ten, this being associated with the use of a geared fan.
- B) The fan blade pitch is variable and hence can be used to produce reverse thrust.
- C) The series of designs included engines with specific provision for large air offtakes and large shaft power offtakes.

Since some of the designs necessitated either a large air or shaft offtake and because the high bypass ratio confers some design advantages it was clearly correct to use the M45 series engines as the basis for the majority of the work.

As far as lift engines were concerned the only possible choice was an engine of the Rolls-Royce RB202 type.

8.2 Conventional Take Off and Landing Designs

It was not thought to be necessary to undertake specific studies for aircraft having conventional take off and landing performance. Adequate information of existing aircraft was available to justify any assumptions necessary and enable the derivation of datum aircraft to be undertaken. Four configurations in this category were used:-

- A) An aircraft with underwing mounted powerplants designed to operate from 5000 ft long runways.
- B) As A but with only a 7000 ft runway requirement.
- C) An aircraft with rear fuselage mounted powerplants designed to operate from 5000 ft long runways.
- D) As C but with only a 7000 ft runway requirement.

The underwing powerplant designs were based on the Boeing 737 series of aircraft. The data for this aircraft was modified to cover the replacement of the existing Pratt and Whitney JT8D engines by the Rolls Royce Snecma M45D. It is worth noting that the Boeing 737/200 series in current service can carry a payload of 108 passengers over a 600 n mile stage length when operating off a 5000 ft long runway. The take off weight in this condition, which corresponds closely with the basic specification stipulated for this investigation, is approximately 90,000 lbs.

The rear fuselage mounted powerplant versions of the conventional aircraft were based on the BAC 1-11 and McDonnell-Douglas DC9 transports. Again a change was made to the higher bypass ratio powerplants.

The 7000 ft runway variants were introduced to furnish extra points in the comparative study, although as can be seen from the quoted performance of the Boeing 737, existing technology has produced aircraft of better field performance than this.

In all cases the modified engine versions of these datum aircraft were found to have a normal economic cruise speed of about $M = 0.7$ at 20,000 ft altitude although the aerodynamic configuration and installed power enabled $M = 0.8$ to be achieved.

8.3 Fan Lift Vertical Take Off and Landing Aircraft

The first of the detailed studies undertaken was for a fan lift vertical take off and landing aircraft. Known as the A 70 the project was used as the subject of the MSc students design work in the 1970/71 academic year. Details of the overall configuration, performance and weights are contained in Appendix D. The full report of the study with some emphasis on the contribution made by the students has been published in reference 61. The configuration of the aircraft was considerably influenced by the earlier VTOL freighter studies described in Section 4.2. For ease of reference the general arrangement is repeated in figure 4. Some features of the design were similar to the Hawker Siddeley HS141 project.

The normal take off weight of the aircraft was 125,000 lbs, and the design landing weight 120,000 lbs. Twelve fan lift engines and two separate propulsion engines were used. The lift engines were derivatives of the Rolls Royce RB 202 type and each was assumed to have a nominal static thrust of 14500 lb giving a total vertical thrust of 174,000 lbs. The implied nominal static thrust/weight ratio of 1.4 included allowances for engine failure, adverse atmospheric conditions and control. The lift engines were housed in two wing nacelles which were located relatively near to the root of the high wing. The two propulsion engines were mounted on either side of the vertical fin. They had a bypass ratio of five and both performance and installation were based on a half thrust scale Rolls Royce RB 211 engine. During transition from vertical to forward flight the thrust of the propulsion engines was augmented by fore and aft tilting of the lift engines.

The wing geometry was chosen to enable the cruise to take place at Mach number of up to about 0.8. A high mounting was used for several reasons:-

- A) The lift engines were well clear of the ground thereby minimising the problems of ground interference and erosion.
- B) No special consideration was given to conventional take off and landing and hence the usual tail clearance problem at incidence near the ground was not present. Thus the high wing could be associated with a relatively low mounted tailplane which was considered to be desirable to give satisfactory dynamic stall characteristics. At the same time the tailplane acted as a partial noise shield between the propulsion engines and the ground below the aircraft during normal climb out from transition.

The use of the high wing did result in the need to locate the lift engines in nacelles away from the fuselage side to reduce exhaust acoustic fatigue difficulties, but not too far out to avoid the difficulties found with the earlier freighter. A further penalty was the small sponsons required to mount the undercarriage off the sides of the fuselage. The take off wing loading of 125 lb/sq ft was determined primarily as a compromise between the conflicting requirements of cruise and transition. Double slotted trailing edge flaps were employed over part of the span to enable the actual transition wing lift coefficient to be set at 1.2, relative to a maximum predicted value of 1.65 in the appropriate configuration. Spoiler/speed brake units were located along the upper surface of the wing, just ahead of the flaps. These were intended for use both at high and transition speeds.

The internal layout of the fuselage is shown in Appendix D, figure 2 and was based on a circular cross section. Six abreast tourist seating was provided for and this arrangement was used as a datum for the other designs studied. The use of a double aisle was found to give an unacceptably large diameter and so a single one was employed. With a seat pitch set at 33 inches it was possible to accommodate 118 passengers. Access to the cabin was by forward side doors and a rear ventral door. The latter was located below the tailplane and a rear mounted auxiliary power unit and was outside the pressure area. Emergency escape exits were positioned above the undercarriage sponsons.

Conventional ailerons, rudder and elevator were used for controlling the aircraft at speeds above transition. Provision was made for trim adjustment of tailplane incidence. During vertical and transition flight control and stabilisation were provided directly by the lift engines. These were arranged as four units of three engines each, port and starboard, and fore and aft. Differential throttling port to starboard and fore and aft was used for roll and pitch control respectively. Yaw control was obtained by differentially tilting the engines port to starboard. The nominal thrust/weight ratio included an allowance of approximately 0.2 for these control functions.

The vertical thrust/weight ratio was selected to enable the aircraft to operate safely at the nominal gross weight from aerodromes up to 5000 ft above sea level and at temperatures of up to ISA + 15°C. More severe conditions necessitated a reduction in take off weight.

A suggested take off procedure was as follows:-

- a) Take off vertically and then climb to 2000 ft at about 15° to the vertical. During this initial flight the climb could be backwards using lift engine tilt for thrust. This would enable a forward descent on to the pad to be made in the event of an emergency. However from a performance aspect it was desirable to gain forward speed in a conventional climb and turn back to the pad if necessary in an emergency.

b) At 2000 ft altitude accelerate in level flight towards the transition conditions. During transition deploy the trailing edge flaps to the intermediate, high lift setting and increase aircraft incidence to give a lift coefficient of 1.2. As forward speed increased tilt the lift engines both to reduce overall lift to the required magnitude and provide a forward thrust component to assist in overcoming intake momentum drag. Subsequently throttle down the lift engines and complete the transition at 190 knots true air speed. This gave a speed margin of 1.2 over the stall speed in the transition configuration.

c) Climb away, retracting flaps, in a conventional manner.

Conventional take off was possible, but the comparatively high wing loading and limited rotation clearance implied the use of a relatively long runway for this type of aircraft.

The normal cruise condition was $M = 0.78$ at 20,000 ft altitude. When a payload of 24,000 lbs was carried the basic still air range with reserve allowance was 550 n miles. The assumed reserve allowance was sufficient for a baulked approach to just above ground level followed by a climb out, 100 n miles diversion and final vertical landing.

The transition to vertical flight at the end of the cruise was carried out at a similar altitude and speed as that following the initial climb. The design landing weight of 120,000 lbs was deliberately chosen to be sufficiently close to the take off weight to enable a safe landing to be made very soon after take off. Transition in this case was initiated by deploying the trailing edge flaps to the high drag position. This gave an adequate speed margin over the stall to start the lift engines at 190 knots true air speed. Initially the lift engines were inclined to give a forward thrust component but subsequently were moved to give first vertical and then an aft thrust component as the nose of the aircraft dropped to reduce aerodynamic lift.

8.4 Tilt Wing Vertical Take Off and Landing Aircraft

A rotorcraft is an alternative way of conferring vertical flight ability. In general the use of rotors throughout the speed range implies a severe restriction in maximum speed potential of the aircraft but it is possible to partially overcome this by tilting the rotors to a vertical plane for the cruise condition. This may be achieved either by tilting the rotor system independently of the rest of the airframe or mounting the rotors on the wing and tilting the assembly. The relative merits of these two possibilities depend to some extent on the role of the aircraft. Previous experience with the small tilt wing executive design described in Section 4.3 suggested that a larger transport version would be feasible and provided a background of data on the concept. As the main reason for considering a rotorcraft was to provide a comparison with the fan lift design it was concluded that the

differences between the tilt rotor and tilt wing would be of secondary importance and the latter was selected because of the background experience.

An initial investigation was undertaken by Martin, reference 62 under the supervision of the author. As much use as possible was made of the work done on the A70 fan lift aircraft and a confirmation of the feasibility of the concept was obtained. Subsequently the design was refined by Ward, reference 63 and used as the students design study for the 1973/4 academic year. The design was known as the R73 and reference to the general arrangement drawings, figure 5, enables comparison to be made with the A70 fan lift study. The layout was very similar to the twin rotor convertiplane projects proposed by Hafner, reference 64 and designs proposed by Westland Helicopters Ltd. Take off weight was 120,000 lbs. The rotor disc loading had to be a compromise between the high disc loading requirements of the cruise and the lower value dictated by hovering. Balancing the power requirements between the two modes for a given cruise speed resulted in a hover disc loading of approximately 36 lb/ft². As distinct from propellers, the rotors were mechanically synchronised to give the aircraft symmetry and contra rotating to balance driving torques. Noise requirements placed a constraint on the rotor performance in the hover. For best hovering performance it can be shown that tip speed should be as high as possible within Mach number limitations but in practice this was limited to 750 ft/sec. The rotors were articulated and the blades were capable of cyclic and collective pitch changes.

The rotors were each driven by two M57 HH turboshaft engines which deliver power to a free turbine. The engine uses the same gas generator as the M45 series fan engine. Due to blade tip Mach number limitations the engine speed needed to be reduced in the cruise to give a rotor tip speed of 550 ft/sec. The flexibility of this type of drive arrangement allowed for almost full engine power to be maintained. In the case of the failure of an engine it was automatically disconnected by a freewheel from the remaining engines, and there was sufficient excess power for the aircraft to land vertically on emergency engine rating in this condition. This was essential since a conventional landing was not possible with this type of aircraft without destruction of the 47 ft diameter rotors, due to the absence of ground clearance when they were in the propulsion mode.

Except for the centre section, the wing was completely immersed in the rotor slipstream. The leading edge was unswept to minimise engine overhang and partly because of this the cruise Mach number was limited to 0.72. During transition the slipstream effect was essential to prevent the wing from stalling. Full span leading edge slats were incorporated with a part span single slotted trailing edge flap. The take off wing loading was approximately 90 lb/ft².

During hovering flight the aircraft was controlled by cyclic and collective changes of the rotors, the total rotor lifting force being 150,000 lbf at take off rating and ISA + 15°C. There was thus sufficient thrust margin for vertical acceleration and control as well as operation 'hot and high'. During transition the conventional aerodynamic controls became effective. The tailplane was mounted at the top of the fin so that the rotor slipstream would have the minimum effect in this phase. Transition was complete at a forward speed of 125 knots TAS. The basic still air range, with allowance for reserves was 540 n miles when 108 passengers were carried. The reserve fuel assumed was sufficient for a baulked approach followed by a 100 n mile diversion.

The cabin arrangement was effectively identical to that proposed for the fan lift aircraft, section 8.4.

8.5 Externally Blown Flap Short Take Off and Landing Aircraft

With the exception of the high pressure internally blown flap system employed in the large STOL freighter described in section 4.1 the concept of externally blown flaps was the first to be investigated in sufficient detail to enable a realistic project study to be undertaken. The exhaust from underwing powerplants is deflected downwards by the trailing edge flaps. The basic configuration and performance of the proposed design is the subject of Cranfield Report Aero No.12, which is to be found in Part 2 of this document. The low speed blown flap characteristics were derived in the main by Ward. The project was designated the A71 and was investigated in detail by the MSc students in the 1971/2 academic year. The results of this study are reported in reference 65. A separate study of the low speed control problems was undertaken by Ward, reference 30. Figure 6 is a general arrangement drawing of the aircraft and the similarity in overall layout to the A70 fan lift study is apparent. Somewhat similar schemes were proposed at about the same time by other investigators in both Europe and North America.

The major layout differences relative to the A70 in addition to the absence of the fan engines may be summarised as:-

- A) The use of four underwing fan propulsion engines.
- B) Double slotted flaps placed in the powerplant exhaust flow.
- C) The provision of a long stroke undercarriage which was wing mounted and housed in wing nacelles.

This last feature was a direct consequence of a design requirement to cater for a steep descent path and an incomplete flare, as discussed in section 7.2. Provision was

also made for castoring the main undercarriage units to cope with the cross wind landing problem arising from the low design approach speed of 79 knots.

As designed the A71 was powered by powerplants of the M45SD type, to give a static thrust ratio of 0.5. The intention was that the aircraft should be capable of operating from 2000 ft long runways and analysis of the landing performance did not reveal any restrictions on this. However in the case of an engine failure on take off the loss of both thrust and lift was such that the take off safety speed required to be nearly 110 knots rather than the 96 knots originally predicted. The corresponding balanced field length was estimated to be 2600 ft. It should be noted that the parametric analysis discussed in section 7.3 and presented in Part 2, Report 18 makes allowance for this effect. The predicted weight of this initial version of the A71 was 115,000 lbs and in this condition the aircraft was capable of carrying 120 passengers over the specified range.

Subsequently the design parameters were revised to ascertain the characteristics necessary to enable the 2000 ft take off condition to be met. It was found that this required an increase of static thrust/weight to 0.68 and a corresponding take off weight increase of some 10%. The initial forward acceleration implied by the high installed thrust would be likely to give rise to passenger comfort problems. However it may be possible to alleviate this effect with little penalty since the main reason for the higher thrust/weight ratio is the need to reduce the loss of lift consequent upon an engine failure at the unstick condition.

The Mach limited cruise at 30000 ft altitude was 0.8 but a more useful condition was rather less than 0.7M at 20,000 ft altitude. This lower cruise speed was determined by cruise comfort conditions resulting from the relatively low take off loading of 74 lb/sq ft.

A problem which soon became apparent with the blown flap configuration was the high noise level due to the scrubbing of the exhaust gases on the flap lower surfaces.

8.6 Augmenter Wing Short Take Off and Landing Aircraft

An alternative means of generating high lift is the augmenter wing arrangement. The major difference between this and the externally blown flap arrangement is that the blowing gases are tapped from the powerplant compressor and expelled rearwards and downwards through a spanwise nozzle formed by splitting the trailing edge flaps into two chordwise elements.

The flap configuration is arranged so that the nozzle air is augmented by the flow over the upper surface of the wing. In one sense the augments wing is a hybrid between the internally and externally blown flap concepts. A particular advantage over these alternatives is the possibility of noise reduction by means of appropriate nozzle design and acoustic treatment in the ducts and the nozzles.

A preliminary study of the effect of replacing the externally blown flaps of the A71 by an augments wing system was made by Van Twisk, reference 66, as an MSc research topic supervised by the author. Superficially the layout was very similar to that of the A71. The major changes made apart from the revised flap system were:-

- A) The use of different powerplants from the M45 series. In this case the engine design allowed for the tapping of very large quantities of air from the compressor. The additional gas generator capacity required to meet this need resulted in an effective thrust increase of about 30%.
- B) The wing mounted undercarriage was replaced by a fuselage mounted one of somewhat shorter stroke. This was done for two reasons, namely to avoid interference with the spanwise extent of the augments flaps and to reduce the undercarriage weight.

The augments nozzle pressure ratio selected was 1.9. It was anticipated that the low speed control problems associated with powerplant failure would be less than those for the externally blown flap A71 due to the possibility of balancing the lift by cross ducting. The greater efficiency of the augments wing and the much higher thrust available relative to the earlier concept resulted in an aircraft which had an estimated runway requirement of only 1600 ft. Landing rather than take off was found to be marginally critical in determining field length. The penalty for these advantages was in the greater all up weight, which was predicted to be about 125,000 lbs and the mechanical complexity of the duct and flap systems.

The augments wing study was continued by Mackichan, reference 32, in conjunction with Ward and the author, reference 67. A more refined design resulted. This substantiated the earlier work in most respects and enabled the low speed control aspects to be investigated. The Boeing Company has also investigated somewhat comparable augments wing transports, reference 29. Use was made of this work to adapt the present design to provide a quieter aircraft, by introducing multi-element nozzles and duct linings.

8.7 Reduced Take Off and Landing Aircraft

The basic A71 design was also used as the starting point for a design which employed only mechanical high lift devices, dispensing altogether with powered lift. The study described in section 7 had indicated that such an aircraft could be designed for fields of no more than 4000 ft length. The preliminary work was the subject of an MSc thesis by Jesse, reference 68 supervised by the author. The configuration is shown in figure 7. The major changes in this case were:-

- A) The use of triple slotted flaps and drooped ailerons rather than double slotted trailing edge flaps.
- B) Reduction of static thrust/weight ratio to 0.46.
- C) The use of a low rather than high wing, which had various layout advantages in the absence of powered lift. A wing loading increase from 71 lb/sq ft to 84 lb/sq ft was found to be possible.
- D) The powerplants were located as a group of three in the rear fuselage. This had one merit in introducing the possibility of using the wing to assist in noise reduction by employing it to shield the intakes.

An alternative version of the aircraft with two underwing powerplants was considered by the author. In both cases the powerplants assumed were of M45S standard.

The take off requirement set the runway length to 4000 ft at a weight of about 104,000 lbs for the rear engined version and about 100,000 lbs for the underwing engine layout. Landing runway requirements proved to be somewhat less than 4000 ft. The higher wing loading relative to the A71 STOL design enabled the normal cruise Mach number to be increased to 0.73 at 20,000 ft altitude for the same standard of passenger comfort.

8.8 Fan Lift Short Take Off and Landing Aircraft

A further method of achieving STOL performance is the use of a number of fan lift engines to provide direct powered lift and thereby augment the wing aerodynamic lift. The layout and concept of such an aircraft is thus intermediate between the conventional configuration and a fan lift VTOL type, unlike the other STOL versions described where the lift augmentation is indirect. It was considered that such a concept would provide valuable data both to compare with the other forms of STOL and to provide a link between the conventional and fan lift VTOL designs. Consequently a design was derived by the simple device of adding six lift fan engines to the rear engined version of the RTOL design. These lift engines were of the RB202 type and were positioned along the lower fuselage sides fore and aft of the wing structural box. During take off it was assumed

that they would be inclined aft to approximately 30° to provide a substantial forward thrust component. The resulting effective static thrust/weight ratio was approximately 0.63. Whilst the design was not regarded as being necessarily an optimum 2000 ft STOL fan lift aircraft it was of interest in being representative of a class of STOL aircraft which could be a direct development from an existing conventional or RTOL type.

8.9 Overwing Blown Flap Short Take Off and Landing Concept

Yet another proposed STOL aircraft configuration is one using powerplants mounted over the wing leading edge with the exhaust flowing over the upper surface of the trailing edge flaps. Coanda effect is used to deflect the exhaust downwards and create a powered lift contribution. The main merit of such a system would seem to be mechanical simplicity relative to the augments wing at the probable expense of some loss of efficiency. At the time the study was being undertaken insufficient performance data had been published to justify a detailed design study. It is of interest that the Boeing YC-14 experimental military transport uses this principle. However it is not anticipated that the advantages of the concept are such as to in any way affect the general conclusions of this investigation although quite obviously overwing blowing must be considered as being competitive with the augments wing.

8.10 Scaling of Design Studies

Although the project studies described had very similar design requirements small differences inevitably arose, primarily as a result of the detail investigations. It was therefore necessary to carry out a scaling process to bring them all to a common base consistent with the basic specification laid down in section 5. This was done on the basis of a weight correction employing the relevant formulae from Appendix B to correct the component weights estimated from the design studies, and using these to build up revised gross weights. No weight scaling was required for the VTOL concepts.

9.0 COMPARISON OF AIRCRAFT CONFIGURATIONS

The comparison of the various design configurations described in the previous section has been reported in Cranfield Report No.24 which can be found in Part 2 of this document.

Whilst the major bases for comparison were the weight, economic and noise properties comments were also made on the relative low speed operating characteristics and fuel requirements.

9.1 Weight

Weight comparison was used in the first instance since a number of other parameters such as first cost and fuel usage, are closely related to it. Particular emphasis was placed upon making realistic allowances for the weights of the flap and powered lift systems. The detail studies were valuable in furnishing data to supplement the more general information contained in Appendix B. The predicted weight breakdowns for the various concepts scaled to a common performance specification are given in Table 2 and figure 5 of Report No.24. A definite trend in weight penalty associated with reduction of field length below 5000 ft was apparent. This amounted to about 12% weight increment for each 1000 ft reduction of runway length down to 2000 ft, below which the rate of increase of the penalty fell markedly. Rear engine designs having the possibility of noise shielding layout proved to be some 3% heavier than a corresponding underwing powerplant layout. Interestingly the 2000 ft fan lift STOL concept was predicted to be some 6% lighter than the blown flap configuration.

9.2 Low Speed Characteristics

Of the STOL designs the augments wing arrangement appeared to suffer less severely than the others from engine failure problems. However any operation from runways below 4000 ft introduced an increasingly more severe low speed problem unless pure VTOL was employed.

9.3 Fuel Requirements

In most instances the fuel requirements were found to be substantially proportional to gross weight, Report 24, figure 8. The exceptions to this were the fan lift designs where the fuel requirements were significantly greater than the norm and the tilt wing VTOL which was predicted to require only 6% more fuel than the 5000 ft runway CTOL design.

9.4 Direct Operating Cost

The trend in direct operating costs was also found to follow closely the pattern set by the weight variations, Report 24, figure 9. Again the exceptions were the fan lift and tilt wing designs. The fan lift STOL was found to have proportionately greater direct operating costs which counteracted the lower comparable weight. The tilt wing design was also found to be relatively much more expensive to operate than both the general trend and the fan lift VTOL. However the cost of fuel was found to be significant in this last comparison and the relative penalty of the tilt wing became much less with increase of fuel cost.

9.5 Noise

The 80 PNdB noise footprints predicted for the design nearly all showed a substantial improvement in comparison with current transports, Report 24, figures 10 and 11. The real exception was the externally blown flap configuration and it is very difficult to visualise ways of significantly improving this. As far as the other designs were concerned the noise trends were unlike those of direct operating cost or weight in that the variation with runway length was much less. A small advantage was shown for the 4000 ft runway RTOL and this was most noticeable for the rear engine, noise shielded layout. The tilt wing aircraft was estimated to be less noisy than the VTOL fan lift design. The augmentor wing concept required to have the full noise treatment to be competitive.

9.6 Overall Comparison

An attempt was made to compare the designs more generally. Two merit indices were introduced. The first was essentially environmental in that it was based on the product of the relative fuel requirement and 80 PNdB noise footprint ratio. On this basis the tilt wing VTOL and designs using 4000 ft or more of runway were very similar but a slight advantage for the rear engined RTOL was apparent, Report 24, figure 12. The second index was primarily economic in that the direct operating cost ratio was used with the 80 PNdB noise footprint ratio. The trend was similar to that of the environmental merit index except that the tilt wing VTOL showed up less favourably. In both cases the externally blown flap and basic augmentor wing designs were shown in a very unfavourable light.

9.7 Conclusions

The major conclusions from these comparisons were:-

- A) Although the rear engine RTOL design suffered some penalty relative to the CTOL aircraft it did possess advantages, especially in regard to the noise levels predicted.

- B) If STOL operations from 2000 ft runways are required the choice is between the quiet augmenter wing and the fan lift designs. Fuel cost is an important consideration.
- C) The tilt wing concept appeared to be very promising in spite of the high direct operating costs due to the mechanical complexity of the arrangement.
- D) The 80 PNdB noise footprints predicted were much smaller in area than those currently experienced but, by and large, were not very dependent upon the length of runway in the specification.

10.0 EFFECT OF AIRCRAFT PARAMETERS ON NOISE FOOTPRINT

One important observation from the noise comparisons discussed in section 9 was the small areas of the 80 PNdB noise footprints in comparison with those associated with current transport aircraft. The various designs considered in the investigation possessed performance characteristics which influenced these noise levels in a complex way. Therefore it was decided to examine the effect of the various aircraft and engine design parameters on the noise footprint to ascertain those where the greatest changes could be expected. This phase of the study has been reported in Cranfield Report Aero No.25 which is included in Part 2 of this document.

10.1 Scope of Investigation

The noise footprints for the different conditions were evaluated in the same way as that employed in the aircraft comparative study. Only the powerplant noise source was considered. The area of the 80 PNdB footprint was used as the reference datum. The parameters considered were:-

- A) Bypass ratio, variation between unity and ten.
- B) Runway field length in the range of 4000 ft to 10,000 ft.
- C) The variation of the rate of noise attenuation in the range of 6.3 dB to 10 dB per doubling of distance from a 500 ft datum point away from the noise source.
- D) The effect of an assumed 5 dB reduction in engine noise resulting from a presumed standard of acoustic treatment or noise shielding.
- E) The difference between a shallow climb out of 10° associated with a shallow descent of 3° in comparison with steep climb and descent angles of twice these magnitudes.
- F) The effect of installed static thrust in the range of 40,000 lb to 160,000 lb.

10.2 Results

The results clearly showed that powerplant bypass ratio was the most important factor for a given installed thrust, although the rate of improvement does reduce markedly as bypass ratio increases, Report 25, figures 1 to 8. This effect alone was sufficient to explain much of the difference noted between those discussed in section 9 and those of existing transports as a bypass ratio of ten was assumed in the majority of design studies.

10.3 Conclusions

The main conclusions were:-

- A) Field length has only a secondary effect on noise footprint area.
- B) The specific thrust of the powerplant is the most important parameter within the control of the designer, but its effect becomes much less significant when the bypass ratio exceeds ten.
- C) Doubling both climb out and descent angle from the values typically in current use is approximately equivalent to a bypass ratio increment of two in the five to ten range.
- D) A 5 dB reduction in the basic noise achieved by acoustic treatment or noise shielding has a similar effect to the doubling of climb and descent angle in low bypass ratio conditions.

11.0 DESIGN PROPOSAL FOR THE NEXT GENERATION SHORT HAUL TRANSPORT AIRCRAFT

The results of the comparative studies have been used to formulate a design proposal for the next generation of short haul transport. This proposal is not directly comparable with the previous designs since an attempt was made to define an aircraft which would meet the requirements of the anticipated market.

11.1 Requirements

The requirements for this design were:-

- A) A passenger capacity approximately 30% higher than that of the existing short haul aircraft, with the possibility of stretch. The majority of the aircraft at present used are in the DC9, Boeing 737, BAC 1-11 category of which some 1400 are in service. These have a capacity of between 100 and 120 passengers in most cases and hence the new proposal should be able to carry about 150 passengers in the initial version.
- B) The comparative study showed some merit in having an aircraft capable of operation from a 4000 ft long runway especially in terms of noise characteristics providing noise shielding is employed. It is worth noting that there are a very large number of aerodromes with runways in the 4000 ft to 6000 ft category in various parts of the world which would be usable. An aircraft with this standard of runway performance can be designed without recourse to powered lift and would not require an extensive research programme such as is associated with a novel feature. There would be a penalty in weight and operating cost but the aircraft would, of course, be capable of operating from longer runways.
- C) In order to have reasonable operational flexibility it was considered that the aircraft should have a maximum cruise Mach No. of 0.8 and be capable of a still air range of the order of 1000 n miles with a capacity payload.
- D) The configuration should take note of the need to reduce fuel consumption as much as possible.
- E) Noise characteristics should be as low as possible with full advantage being taken of noise shielding, high bypass ratio and steepening of climb and descent as far as was practicable.

- F) The basic layout should be such that in the event of suitable fan lift engines becoming available it would be possible to develop the aircraft first as fan lift STOL and then as a fan VTOL aircraft.

Summarising, the concept was to be for a quiet 4000 ft RTOL aircraft capable of carrying about 150 passengers over about 1000 n miles at a maximum cruise speed of $M = 0.8$. Developed versions could either be for about 180 passengers from longer runways or, by the addition of lift fans, STOL or VTOL aircraft.

11.2 Configuration and Performance

The design produced to meet these requirements was designated the A74 and used as the basis of the students project work in the 1974/5 academic year. The geometry, aerodynamic, performance and loading data are contained in Appendix E.

A general arrangement drawing is shown in figure 8. The layout was unusual in two main respects:-

- A) Effectively unswept wing with a relatively high aspect ratio of 9.2. This was made possible by using a 0.13 thick supercritical aerofoil section. The wing geometry was chosen to give good lift/drag ratio under take off the cruise conditions as well as an essentially unswept trailing edge flap system. The supercritical aerofoil represented a departure from the earlier designs but it was deemed to be a reasonable provision in view of the standard of technology now achieved in this connection.
- B) Engine location, which was on the fuselage between the wing and tail. The arrangement was selected to gain the maximum noise shielding effect from the wing and tail surfaces.

As proposed the aircraft was powered by four fan engines of the Rolls-Royce-Snecma M45SD type similar to those of the previous studies. These have a bypass ratio of approximately ten and the assumed static thrust was 14500 lb each. Four powerplants were provided for in the design simply because of the absence of data on larger units of comparable performance. In practice it is envisaged that the installation would consist of two engines, each of approximately 30,000 lb static thrust.

For RTOL operations the take off weight of 130,000 lb implied a static thrust/weight ratio of 0.45.

The low wing had a slightly swept leading edge, which was kinked part of the way along the span. The greater sweep at the root was primarily for purpose of enabling the structural box to be cranked forward at the fuselage. This allowed the four wheel bogie main undercarriage units to be retracted into

the lower fuselage aft of the wing box but even so the provision of structure for the reaction of drag loads was found to be a problem.

The wing area of 1460 sq ft corresponded to a take off wing loading of 89 lb/ft^2 and together with the relatively high aspect ratio resulted in a wing structure designed by gust cases. The high lift devices consisted of triple slotted trailing edge flaps over about 70% of the span and leading edge slats outboard to prevent premature tip stall. These enabled an approach lift coefficient of two to be used which resulted in an approach speed of 107 knots at a landing weight of 115,000 lb approximately.

The tailplane also used a supercritical aerofoil section and since the cruise trim load was normally down the section was inverted. Longitudinal control was from the tailplane/elevator combination, whilst conventional ailerons and rudder were used for lateral and directional control. Spoilers were located along the upper wing surface over the flaps. It was intended that these be used as air brakes, lift dumpers, and for direct lift control.

The passengers were accommodated in six abreast seating with a single central aisle. The initial design provided for a total of 146 tourist class seats at a minimum seat pitch of 31 ins. Access to the cabin was through side doors at the front and a ventral door at the rear. Airstairs were provided for rapid turnaround.

The maximum cruise Mach number was predicted to be 0.83 at altitudes between 20,000 ft and 30,000 ft. The range performance depended upon the flight pattern employed. For short flights at a cruise altitude of 20,000 ft the range with full payload was the equivalent of 800 n miles in still air with no reserve allowance, and a cruise condition of $M = 0.8$. A slower cruise speed of $M = 0.6$ enabled this to be increased to about 1150 n miles. In the case of longer flights where the preferable cruise altitude was 30,000 ft the corresponding ranges at cruise speeds of $M = 0.8$ and $M = 0.6$ were predicted to be 1150 n miles and 1300 n miles respectively. When allowance was made for reserves the latter implied an operating range of about 860 n miles.

12. CONCLUSIONS

- (1) Passenger comfort considerations have a major bearing on the design characteristics of aircraft intended for short runway operation. A direct result of this is the need to use some form of power augmented lift when genuine STOL performance from 2000 ft runways is the requirement.. This together with the implied low take off and landing speeds results in many design, development and operational difficulties.

The augments wing and overwing blowing concepts present the most promising possibilities for the solution of these problems although a fan lift STOL aircraft is worth consideration if fuel costs are not of overriding importance. There is considerable doubt as to the need for an aircraft in this class.

- (2) On the other hand the development of an RTOL aircraft for operation from 4000 ft runways is well within the current state of technology. Only mechanical high lift devices are required. Such an aircraft would suffer some performance and cost penalties relative to an aircraft intended for conventional runway operation but could be designed to use longer runways competitively. It offers the best opportunity for noise reduction and could also form the basis of fan lift STOL and VTOL aircraft should the appropriate powerplant become available and there be a demand for this type of vehicle.
- (3) Although a VTOL aircraft suffers some economic penalty relative to the other classes of aircraft studied it does have the advantage of eliminating some of the low speed control problems associated with STOL. The tilt wing/rotor concept for VTOL appears to offer some potential gains in terms of noise and fuel usage relative to fan lift and in spite of the mechanical complexity these make it worthy of consideration.
- (4) The design field length has only a secondary effect on the noise footprint. Specific thrust of the powerplant is the most important parameter. Airframe self noise can be an important consideration when engine bypass ratio is of the order of ten and maximum use is made of acoustic treatment and noise shielding layout.
- (5) A possible next generation short haul transport intended to replace the majority of existing types is one based on a 4000 ft RTOL concept carrying about 150 passengers and using a noise shielded layout. A design to meet these requirements is presented.

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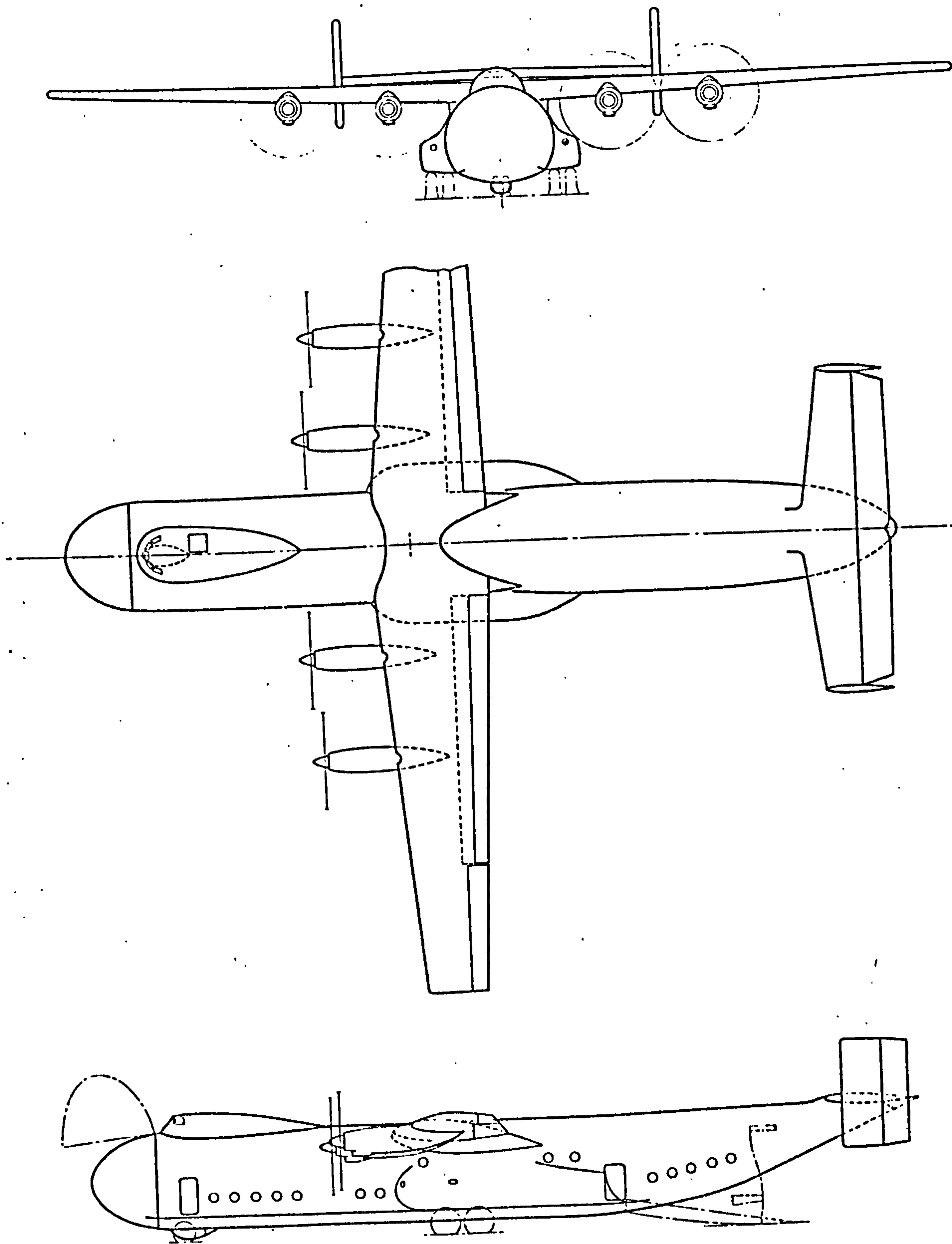
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DEPARTMENT OF AIRCRAFT DESIGN.	
GENERAL ARRANGEMENT - 1959 DESIGN PROJECT (TACTICAL AND STRATEGIC FREIGHTER - F 59)	
SCALE. - $\frac{1}{100}$ (8 1/2 to 11 in)	
DRAWN. - D. HOWE. 10/7/59	F 59-1

FIGURE 1. GENERAL ARRANGEMENT OF F59 DESIGN

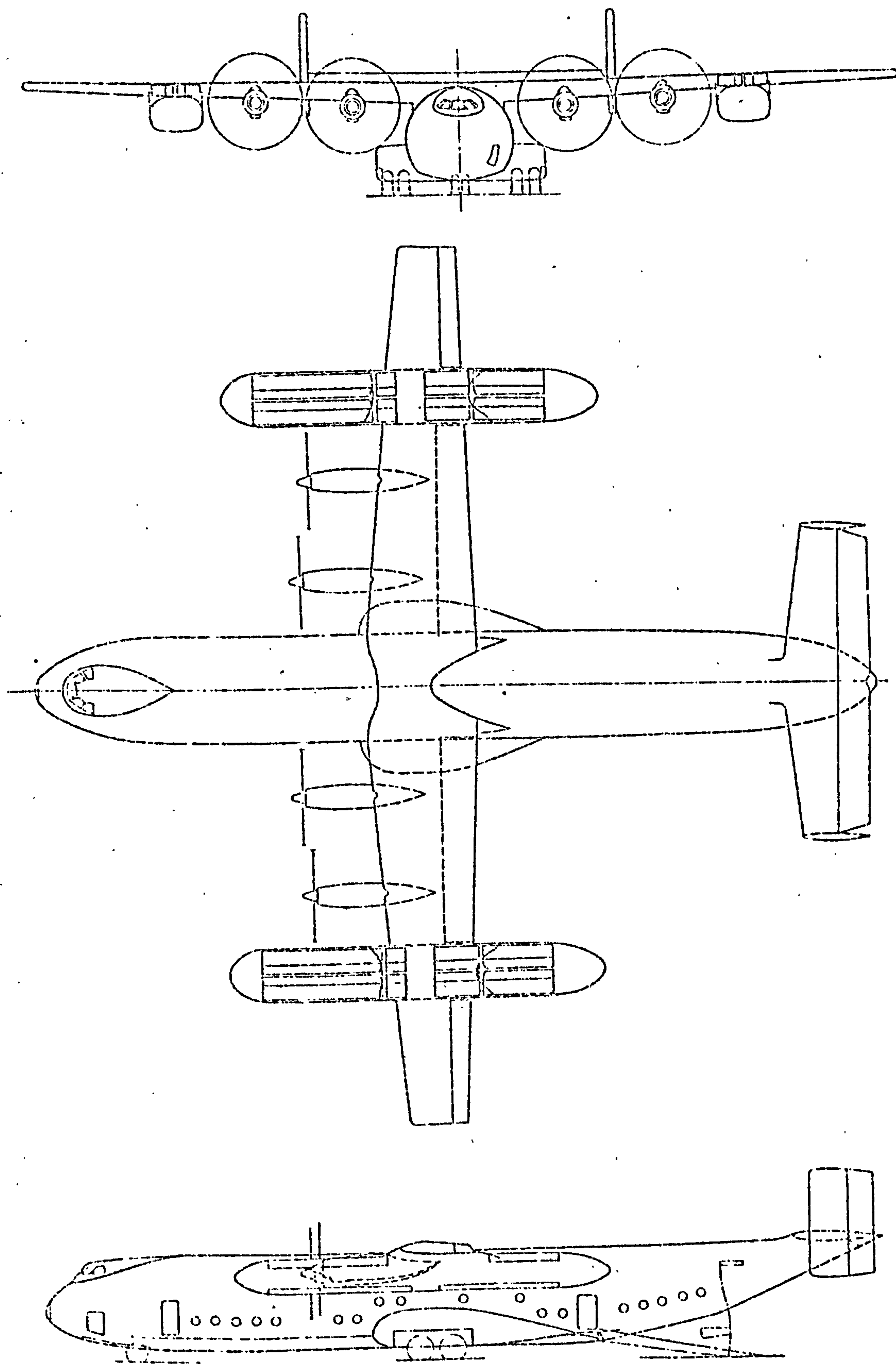


FIGURE 2. GENERAL ARRANGEMENT OF F61 DESIGN.

DEPARTMENT OF AIRCRAFT DESIGN	
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DRAWN -	
B. HONE 10/1/61	F 61-1

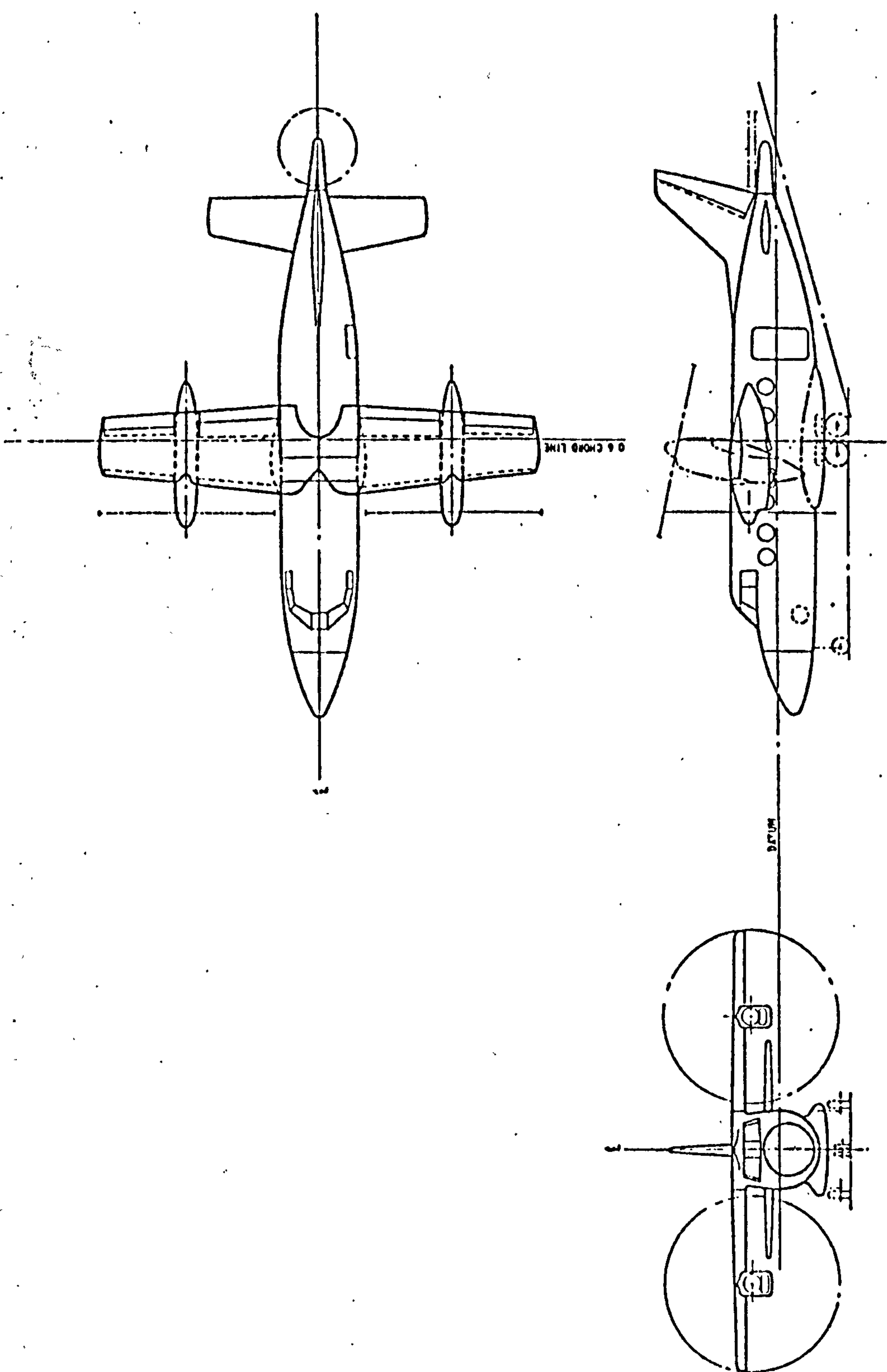
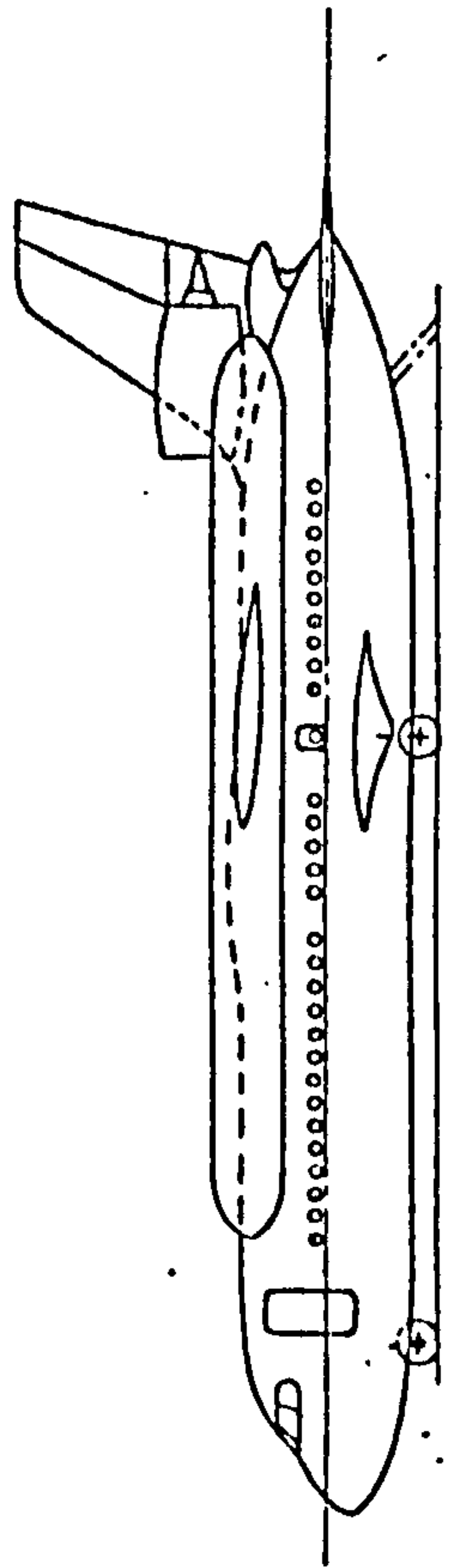
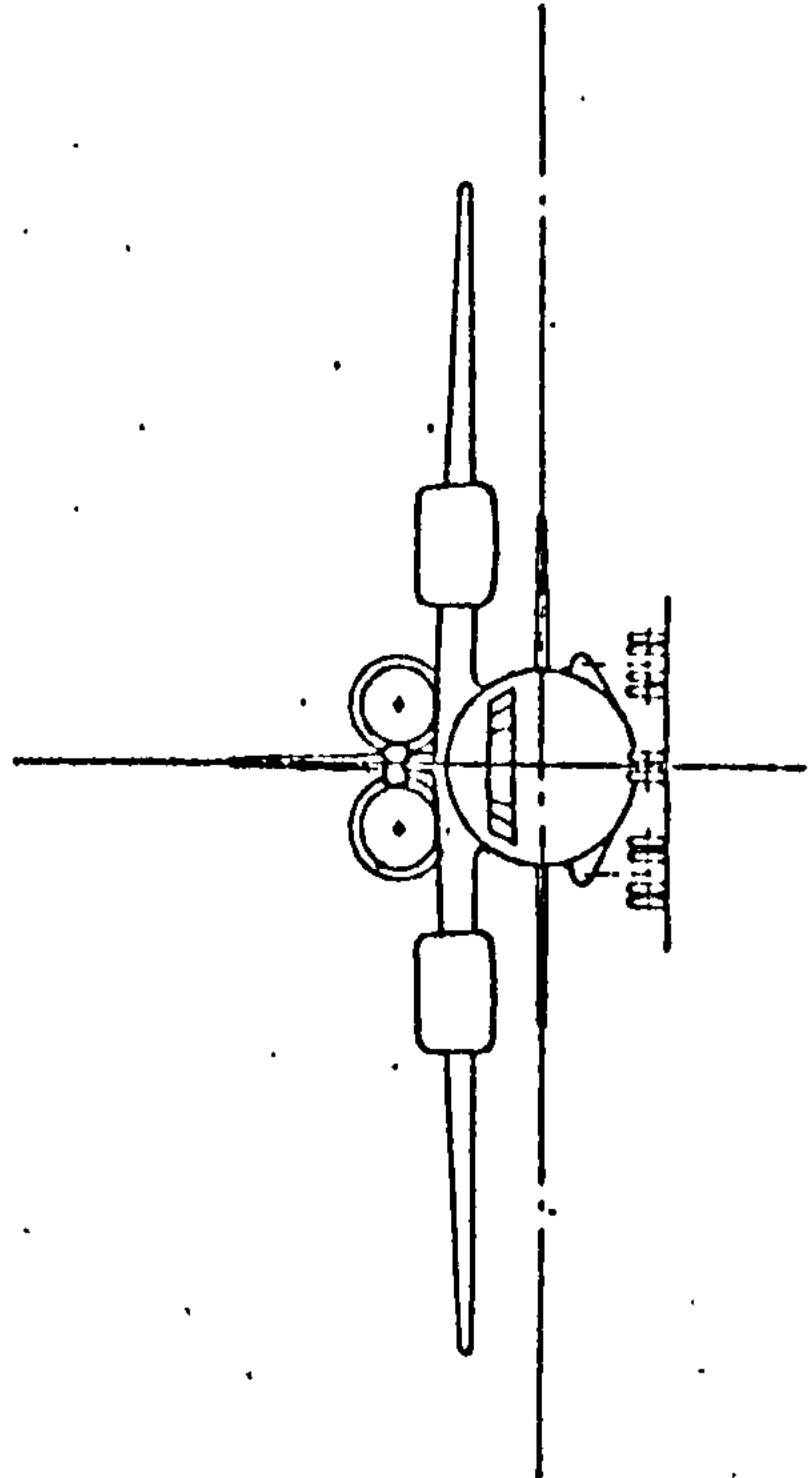
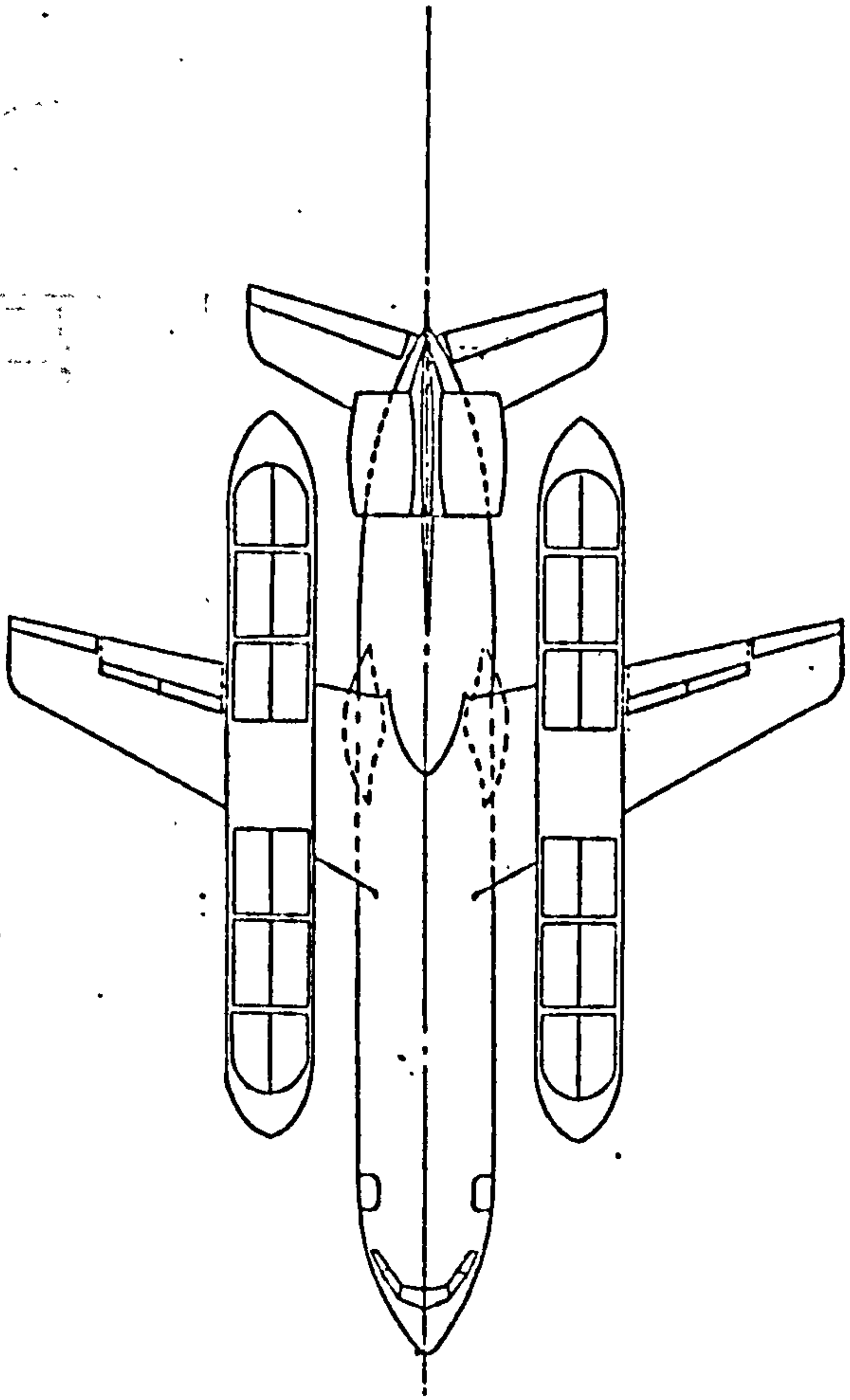
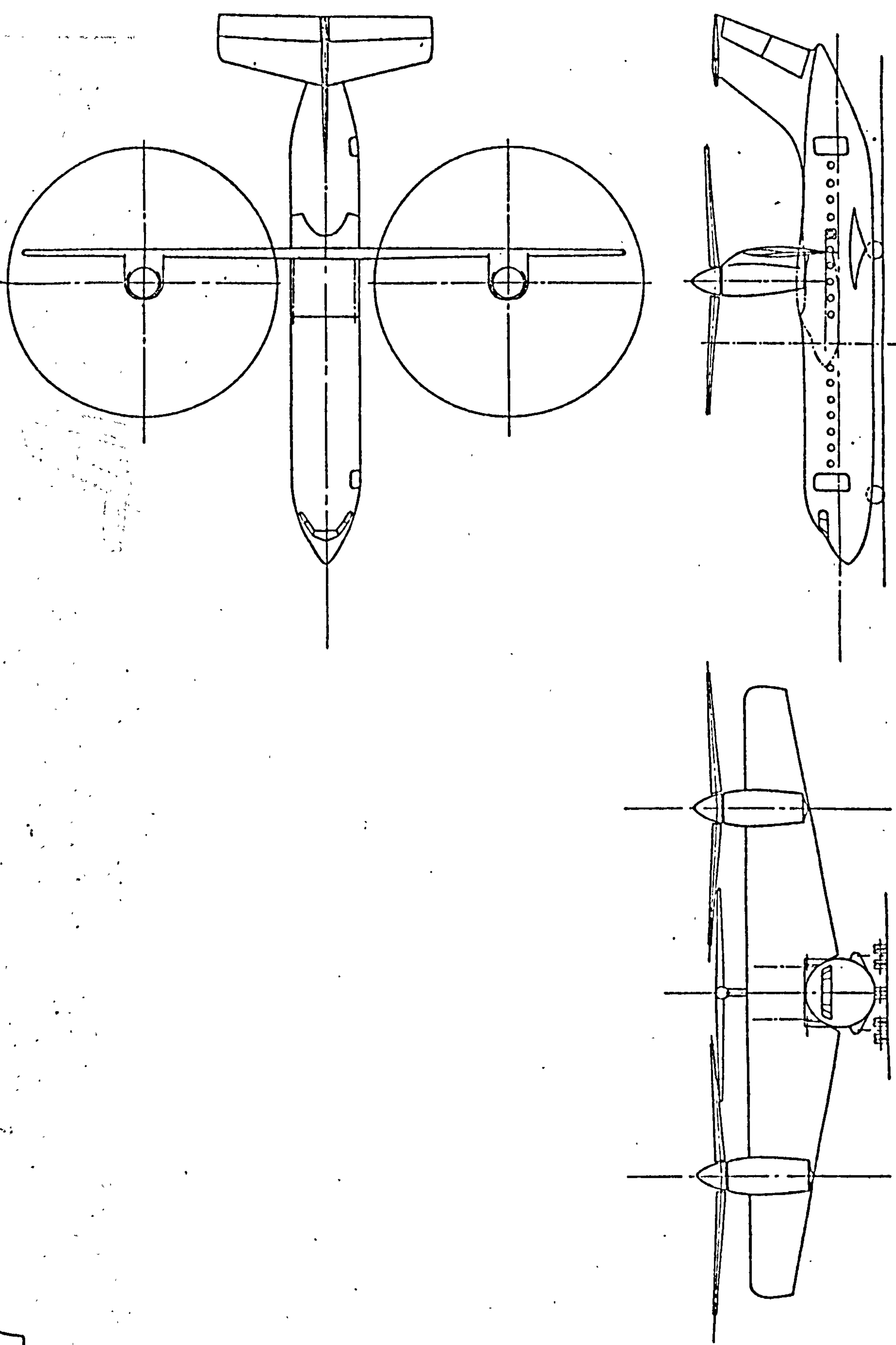


FIG. 3 GENERAL ARRANGEMENT OF E67 DESIGN.



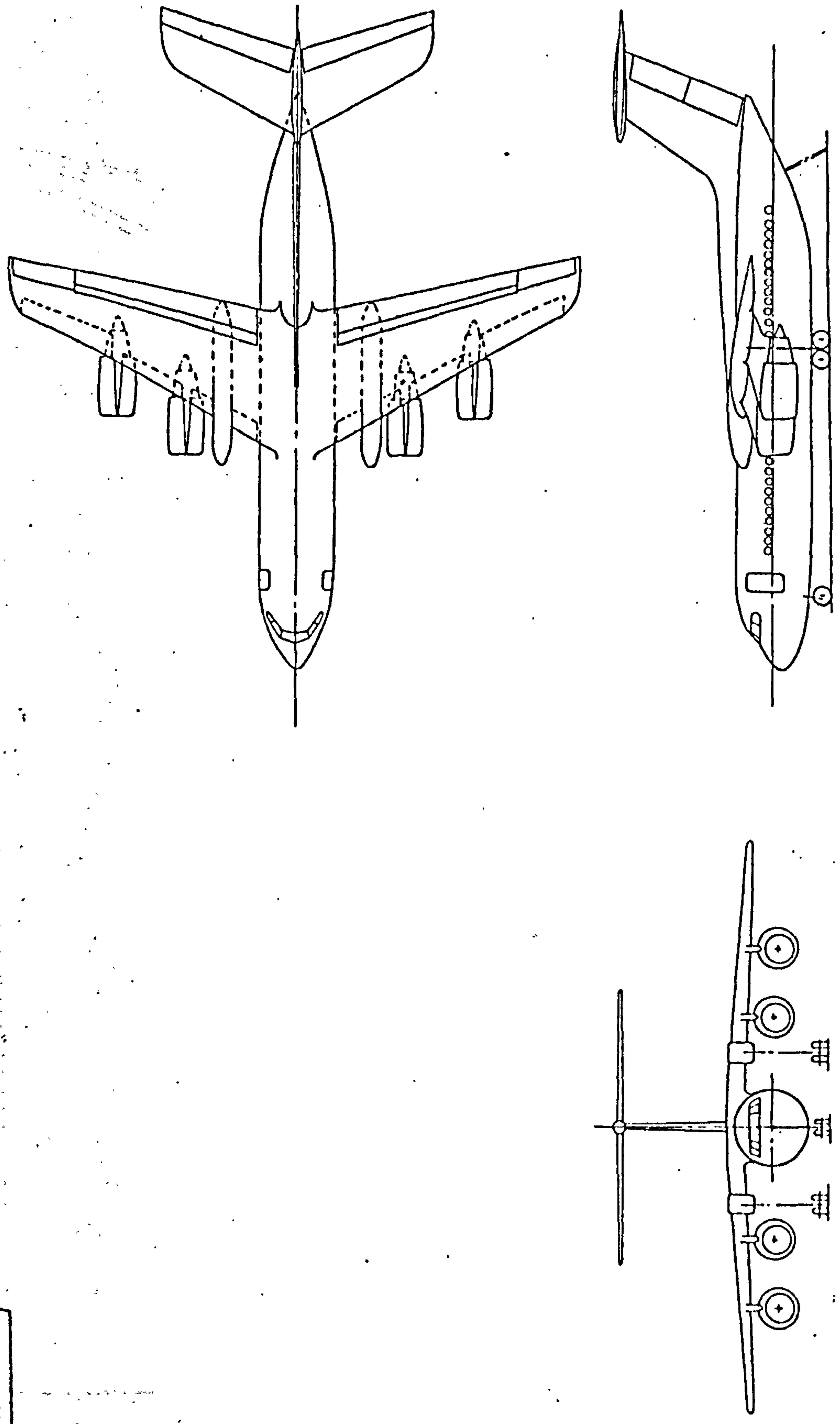
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GENERAL ARRANGEMENT			
DRAWN	DATE	DESIGNED BY	NO.
J. H. H.	1/1/70	A70-1	1
SHEET		OF	
1		1	

FIG. 4 GENERAL ARRANGEMENT OF A70 FAN-LIFT VTOL.



TITLE: 1973 PROJECT	
GENERAL ARRANGEMENT	
SCALE: ~	ISSUE: ~
DRAWN: ~	
DATE: 16/7/73	R 73-1

FIG. 5 GENERAL ARRANGEMENT OF R73 TILT WING VTOL



TITLE - 1971 DESIGN PROJECT			
GENERAL ARRANGEMENT			
SCALE - 1/600	ISSUE - 1		
DRAWN - S. H.	A 71 - 1		
DATE - 29/7/71			

FIG. 6 GENERAL ARRANGEMENT OF A71 EXTERNALLY BLOWN FLAP STOL.

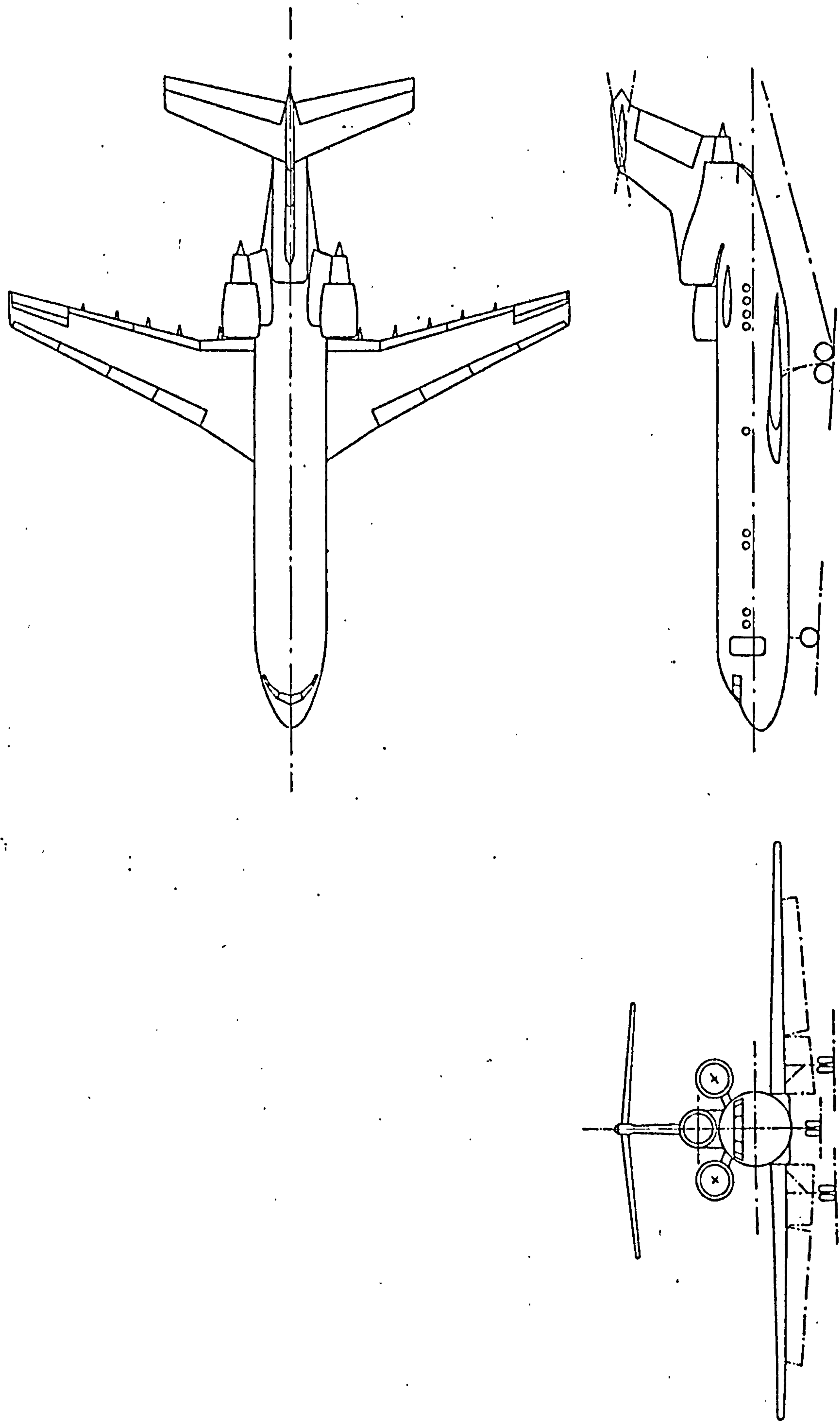


FIG.7 GENERAL ARRANGEMENT OF REAR ENGINE RTOL.

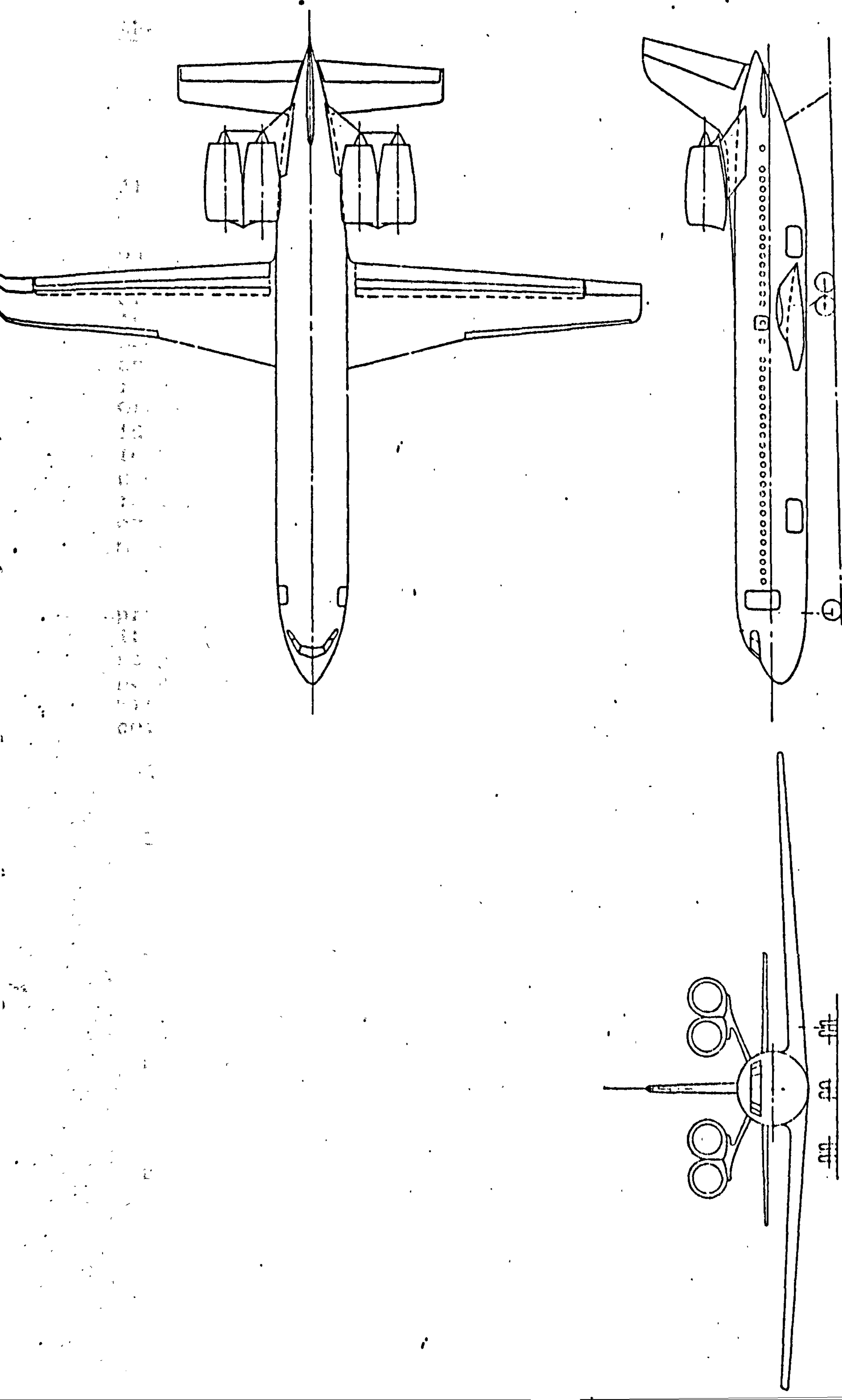


FIGURE 8. GENERAL ARRANGEMENT OF A74 RTOL AIRCRAFT

DRAWN	CHE	APPV	SCALE	TITLE:-	A-74
14 8/74			1/100	GENERAL ARRANGEMENT	
COLLEGE OF AERONAUTICS				DRAWING NO	A74 -1
CRANFIELD INSTITUTE OF TECHNOLOGY				3rd	CF SHEET
CRANFIELD					

APPENDIX A

Basis of Noise Comparisons

A1 INTRODUCTION

Various requirements have been added to the airworthiness ones required for the certification of civil aircraft in an attempt to control the aircraft noise nuisance. The earliest of these were those specified by the United States Federal Aviation Authority. The requirements of FAR, Part 36 are somewhat complex in detail since they are intended to be applied in the form of experimental noise measurements. It is necessary to provide corrections for wind, temperature, etc. Other authorities have followed suit with similar but not identical requirements. Examples of these are Annex 16 to the ICAO requirements and Section N of British Civil Airworthiness Requirements. In effect the application of these requirements is limited to conventional subsonic transport but suggestions or recommendations for other classes of aircraft have been made.

The FAR Part 36 requirements are typical of those at present in existence. In order to comment effectively on them it is necessary to summarise the basis on which they are stipulated and applied. The FAR requirements state three points relative to the aircraft flight path at which compliance with the specified noise limits must be demonstrated. These noise measuring points are:-

- A) 3.5 n.miles from the start of the take off roll, on the extended runway centreline.
- B) On a sideline 0.25 n.miles from the runway centreline where the noise is greatest at take off, except that if the aircraft has more than three engines the distance is 0.35 n.miles.
- C) On the approach, 1 n.mile from the threshold on the extended centreline of the runway.

The actual noise levels allowed are:-

- a) For the take off case, A) above, 93 EPNdB for gross aircraft weights up to 75,000 lbs, rising to 108 EPNdB at 600,000 lbs. The increment is 5 EPNdB for each doubling of the weight.
- b) For the sideline and approach cases, B) and C) above 102 EPNdB at 75,000 lbs rising to 108 EPNdB at 600,000 lbs weight. In this case the increment is 2 EPNdB for each doubling of the weight.

The unit used for noise measurement is the effective perceived noise level, EPNdB. The process of deriving this unit is:-

- (i) Correct measured sound pressure level for ambient temperature, pressure, humidity and wind to the reference conditions.
- (ii) Adjust the correct value to allow for deviations of the flight path from the standard specified.
- (iii) Convert the one-third octave bands to perceived noisiness in noys, and thence to instantaneous perceived noise levels.
- (iv) add a correction factor to allow for discrete tones, etc. using the noise measurements at each half second interval during the fly over.
- (v) Determine the maximum value of the tone corrected noise levels and then add a correction to allow for the duration of the fly over. This is effectively an averaging process and gives the noise level in EPNdB.

In common with other existing requirements FAR Pt.36 allows engine power to be reduced before the aircraft passes the take off measuring point and flight path changes to be made. Therefore in some instances operators have been able to meet the regulations with aircraft which were basically noisier than allowed.

Since the certification requirements apply only to an individual aircraft of a given type they are not a measure of the total noise nuisance of all the aircraft operating from a given airport. Various systems have been introduced to cover this case. For example in the United Kingdom it is common for total noise nuisance to be based on the Noise and Number Index, or NNI value. The NNI value at a given location near and airport is derived from an averaging process of the number of operations of different types of aircraft of given noise characteristics. On a somewhat similar basis in the United States an important proposal is one which introduces a so called Fleet Noise Level, or FNL number. Each operator from a particular airport would be allocated an FNL for his total operation, based on the following formula:-

$$FNL = 10 \log_{10} \left[\frac{\sum N_i \text{antilog} L_i / 10}{\sum N_i} \right]$$

where N_i is the number of operations in a 90 day period of aircraft of type i .

L_i is the noise level in EPNdB of one aircraft of type i .

The noise level would be evaluated both on take off and approach. Once established and agreed the operator would not be able to exceed the value of FNL, but he would be able to change aircraft type, number of operations, etc. within the limit. This is obviously an encouragement to use new aircraft or fit noise suppression kits to reduce the values of L_1 in an expanding situation, or retire older, noisier types.

Elaborations of this approach to the problem are possible. One example of this is the work of Richards and Ollerhead, Reference 10, which enables an assessment to be made on overall basis so that the conditions at different airports may be compared.

Whilst it is possible to compare the noise levels of projected aircraft at points corresponding to those specified in the certification requirements this can become somewhat complex. Commonly the simpler approach of comparing noise levels at, say, 500 ft from the source is used. Alternatively an attempt is made to evaluate the total ground area below and around the aircraft during the take off and landing cycle which is subjected to a given noise level. This so called 'noise footprint' approach has often been based on the 80 or 90 PNdB contours. Again it is possible to use more elaborate techniques for example Kalk, Reference 11, has introduced the idea of a Specific Annoyance Factor or SAF. This attempts to compare the total annoyance created by a given aircraft on a given flight in a particular airport situation and is especially valuable in ascertaining the true effect of noise abatement procedures.

A2. COMMENTS ON NOISE REQUIREMENTS AND FUTURE TRENDS

An important aspect of the existing noise requirements is the obvious influence of the aircraft manufacturer and operator. This is most apparent in the alleviation in noise level allowed for large aircraft and in the concept of the Fleet Noise Level. There is no reason at all why a person on the ground below the flight path should accept a higher noise level because the aircraft happens to be heavier, or why he should accept being woken up at 15 minute intervals during the night rather than 10 minute ones. The allowance for large aircraft cannot even be justified on economic grounds since large aircraft cost less to operate anyway. The existing requirements are perhaps best regarded as a device introduced to ensure that the trend of increasing aircraft noise nuisance was reversed. This has obviously been successful but the suggestion already made that Part 36 noise levels should be reduced by 10 EPNdB is evidence that much more severe restrictions will be imposed in the future. Indeed there have already been court decisions in the United States as a result of which curfews have been imposed upon aircraft which meet the present requirements of Part 36. There is no doubt that because designers have already

produced quieter aircraft the general public will expect each new type of aircraft to be less noisy than its predecessors. This demand will only cease when the situation is reached that aircraft noise is sensibly within that of the local environment.

The use of a noise criterion based on this ultimate concept does introduce certain difficulties. In the first place the economic penalty of reducing the noise of the aircraft to the value required must be assessed and accepted. The magnitude of the penalty is dependent upon the type of aircraft and many details of its design.

Secondly it is necessary to establish the target noise level and the method of defining it. This has been the subject of a number of investigations, and again is a function of time and location. In the case of a typical urban environment Clarkson, Reference 12 has suggested that the aim should be to hold the background noise level to about 70 dB(A). A study by Lilley, Reference 13 concludes that the level should be somewhat lower than this, probably in the range of 55 dB(A) to 65 dB(A). The subjectivity effect covered by the conversion to perceived noise level is of the order of 10 dB to 14 dB in the case of aircraft noise so that 66 dB(A) is approximately equivalent to 80 PNdB. Although this may be somewhat high on the basis of Lilley's findings it is a convenient level to use for aircraft comparison purposes in the context of the noise footprint. It is reasonable to deduce that people in an urban daytime environment will, typically, be aware of the presence of the aircraft but are not likely to be at all annoyed by it. The presence of the aircraft would be more noticeable at night or in a rural environment and to cater for this case the 70 PNdB contour might be more appropriate. However whereas there is some prospect of designing aircraft which have noise characteristics such that the 80 PNdB footprint is contained largely within airport boundaries, there is virtually no prospect of this being achieved for the 70 PNdB footprint.

Some idea of the implications of meeting the 80 PNdB criterion at the FAR Part 36 measuring points can be gained by reference to Figures A1 and A2. Although these are not directly comparable with the 80 PNdB footprint area the severe problems associated with large aircraft is obvious. It is possible that noise considerations may provide one factor which determines the ultimate size of transport aircraft. Aircraft intended for reduced, short or vertical take off and landing also face a relatively severe problem due to the higher installed thrust which is implied by the performance requirements. Fortunately in this case alleviation of varying degree may be possible because of their low speed potential.

The concepts of Noise and Number Index and Fleet Noise Level become irrelevant if the noise of an individual aircraft is no greater than that of the general background.

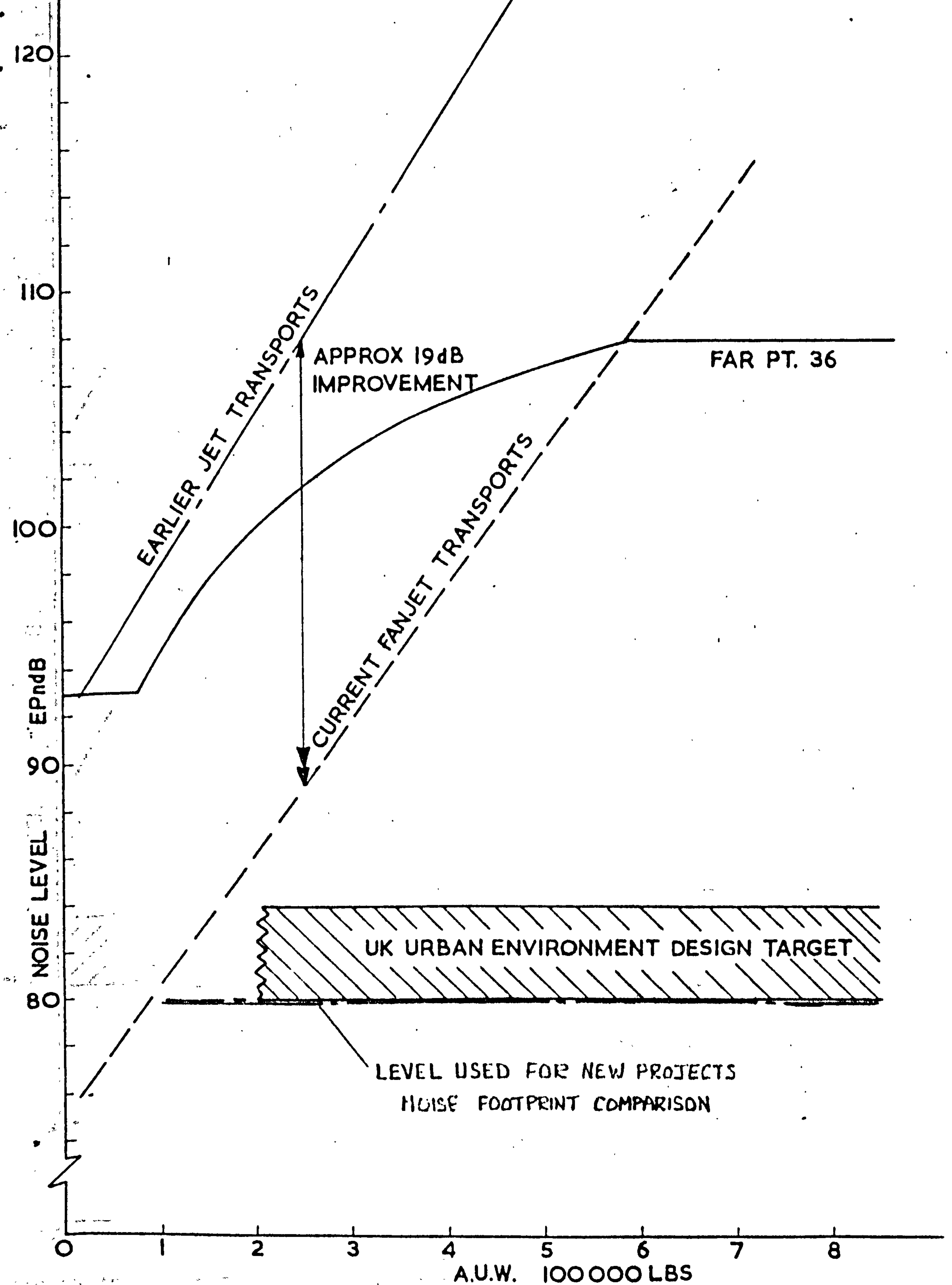


FIGURE A1.

TAKE OFF NOISE LEVEL

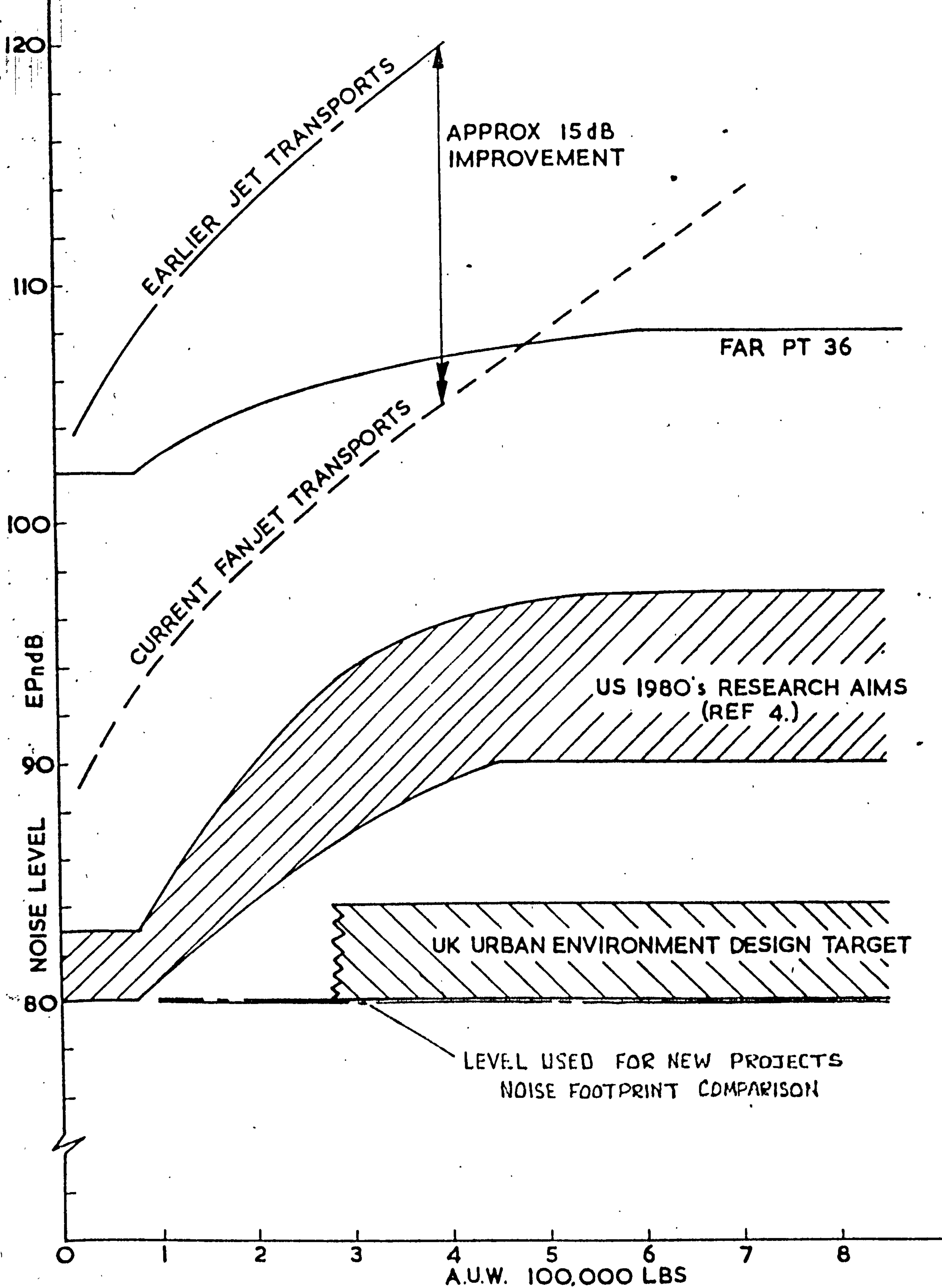


FIGURE A2. APPROACH NOISE LEVELS

APPENDIX B

Aircraft Weight Prediction

(This appendix is based on Lectures Notes DES 126 and 129 prepared by the author)

B1. INTRODUCTION

All aircraft weight prediction depends for accuracy upon the skill and experience of the designer. Various approaches to the problem are possible, depending upon the degree of complexity which is desirable.

a) Empirical comparisons.

In some limited instances it is possible to make approximate predictions by direct comparison with known existing designs. In all cases it is desirable to check predictions against figures for similar aircraft and hence an important part of the weight prediction process is the collection and collation of weight data. Experience plays an important part in interpretation of this data since all designs differ in certain respects.

b) Empirical formulae

An extension of the direct comparison approach is the use of formulae which have been derived directly from known weight breakdowns. The derivation of such formulae may make no attempt to interpret results on a theoretical basis, but quite obviously they are of more use if some attempt is made to present them in terms of parameters suggested by theory. These formulae are usually simple and require only a relatively small amount of initial information. They are particularly suitable for initial design work when it is desired to carry out wide ranging parametric studies.

c) Theoretically derived formula

It is possible to derive weight prediction formulae by means of a theoretical design approach. The major difficulties in this case arise from the simplification necessary to ensure that the resulting formula is manageable and the need for extensive empirical corrections to cover practical design considerations. It is worth noting here that experience suggests that there is a limit to the usefulness of such formulae and attempts to refine them by more careful and detailed theory often give less satisfactory results. The main use for such formulae is for more detailed parametric comparisons of geometry variation, etc.

d) Prediction methods

Where greater accuracy is required than is obtainable by the use of formulae it is necessary to employ a prediction method. Such techniques imply a preliminary design process, again corrected by empirical information. Prediction methods are well suited to computational techniques and can enable rapid and accurate weight figures to be obtained. As a design is refined the prediction becomes an estimate based on known details. Prediction methods are especially valuable when a design is being developed from an existing basis since in this case the empirical factors are well established and changes in layout or loading can readily be accommodated. Although rapid results are obtained by the use of computers the storage and input data necessary may present difficulties in an overall preliminary design process.

The structure of an aircraft accounts for about half of the empty weight and is the portion most directly within the control of the designer. For this reason much of the effort put into weight prediction techniques has been devoted to the structural aspect.

Great care is necessary in interpreting the stated weight breakdown of a given aircraft and comparing it with prediction formulae. It may be just as incorrect to conclude that a discrepancy is because the formula is in error as to deduce that a design is inefficient because it appears to be heavy relative to another apparently identical design. Although standard forms are used to state weight breakdowns individual interpretations vary widely. Weights are often allocated in a way which is convenient to checking during manufacture of assemblies. For example it is rarely immediately obvious just how wing-fuselage junction weight should be stated. Even total structure weight can be misleading as it may include differing amounts for mounting brackets, systems items, etc. Operating empty weight can be a much better guide to design efficiency.

B2. Wing structure weight

The wing lends itself to weight analysis because of the well defined structural role. Typically the wing accounts for one tenth of the gross weight but large variations occur in particular cases. The most important parameters are the wing geometry and thickness, the design weight and normal acceleration factor and the design diving speed.

Analysis of known wing weights of some 100 recent aircraft of all types suggests the following relationship:-

$$W_W = C_1 \left[\frac{bS}{\cos \phi} \left(\frac{1+2\lambda}{3+3\lambda} \right) \left(\frac{WN}{S} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \text{ lbs} \quad \dots (1)$$

where b is the wing span (ft)

S is the wing area (sq ft)

ϕ is the quarter chord sweep angle

τ is the thickness/chord ratio at the root.

λ is the ratio of the tip to root chords

W is the design weight (lbs)

N is the factored design normal acceleration factor

V_D is the design diving speed (knots)

C_1 is a coefficient which varies according to the type of aircraft and layout details. Typical values of C are usually within the range 0.0027 to 0.0037. More specifically:-

Long range aircraft: $C_1 = 0.0029$

Short range transports $C_1 = 0.0035$

Light aircraft $C_1 = 0.0029 - 0.0035$

according to complexity and in particular flaps and cutouts for retracting undercarriages, etc. $C_1 = 0.0021$ if the wing is braced rather than cantilevered.

Fighters and fighter/bombers $C = 0.0029 - 0.0035$ with an increment of about 0.0005 for naval aircraft.

When the wing geometry is not greatly variable it is possible to use simpler relationships. For example the taper and aspect ratio of delta wings does not vary a great deal, nor does the (V_D/τ) ratio in usual applications and in this case:-

$$W_W = 1.24S^{1.06} \left(\frac{W}{S}\right)^{0.3} \text{ lbs} \quad \dots (2)$$

Jet transport aircraft with an aspect ratio of about 7 and typical taper ratio about 0.3 give:-

$$W_W = 0.028W^{1.1} \text{ lbs} \quad \dots (3)$$

A much more elaborate formula based on a theoretical analysis is:-

$$W_W = D + \left(E + q + \frac{7.2m}{\bar{M}W^{0.09}}\right) W \times 10^{-3} + \left[3k \frac{W}{S} S_f \times 10^{-3} + 0.1 \left\{ (1-e)S - F_f \right\}\right] V_D^{1/2} \\ + 0.6Se \left[(1+\lambda)(c\tau)^{1/2} - 1.2(c\tau)^{-1/2} \right] \text{ lbs} \quad \dots (4)$$

where D is the greater of D_B , designed by bending considerations and D_T , designed by torsional stiffness criteria. * last term should be replaced by

$$\text{or } 0.9Se(1+0.55\lambda) \left[(c\tau)^{1/2} - 0.65(c\tau)^{-1/2} \right] \\ \text{for root depth} > 0.75 \text{ ft.}$$

$$D_B = 1.3WNR(1+\lambda)b\sec\gamma \times 10^{-5} \left[1 + \frac{20.7(1+\lambda)(1-0.44\lambda^{\frac{1}{2}})A\sec\phi}{\{100\tau+3.55(1+\lambda)NWrA\sec\phi\sec\gamma \times 10^{-5}/Lec\}} \right]$$

The first term in the brackets is the weight due to shear webs and the second bending material, the second term in the denominator allowing for the higher working stress possible with higher end loads.

$$D_T = \frac{4b^3}{\cos\gamma}(1+3.08\lambda)\left(\frac{1+1.79\tau}{e\cos\gamma}\right)\left\{\frac{V_D g \cos^{3/2}(\Lambda-11^\circ)}{\tau(1-0.166M\cos\Lambda)}\right\}^2 \times 10^{-10}$$

$$L = (ct)^{\frac{1}{2}}$$

S_F is the total flap and slat area (sq ft)

F_f is the part of the wing area, S , accounted for by the flaps and slats on both upper and lower wing surfaces (sq ft)

E number of engines mounted on wing

e is fraction of wing chord occupied by structural box

c is the wing root chord (ft)

m is the number of main undercarriage units mounted on wing out of a total of \bar{M} main undercarriage units

A is aspect ratio

M is the Mach number corresponding to V_D

g is the fraction of the chord aft of the leading edge to the inertia axis, but not less than 0.4

γ is the sweep of the structural box

Λ is the sweep of the leading edge

k is the flap factor

k is 1.0 to 1.5 for trailing edge flap according to complexity.

1.5 for leading edge flaps

2.0 for leading edge slats

r is the wing bending relief factor; $r = (1 - \frac{2R}{W})$

where R is the effective relief load/side acting at 40% semispan.

l rib pitch.

For project work r may be evaluated as $1 - \sum \Delta r$ where:-

$\Delta r = 0.1$ for structure and systems weight

0.03 for two wing mounted jet engines

0.05 for two wing mounted turboprop engines

0.10 for two wing mounted piston or four
wing mounted jet engines

0.15 for four wing mounted turboprop engines

0.20 for four wing mounted piston engines

$0.6 \left(\frac{W_T - W_L}{W_T} \right)$ for fuel weight

where W_T and W_L are the design take off and landing weights respectively.

q is a factor determined by the structural penalties and efficiency of the wing design. Some experience is desirable in choosing values for this but as a guide:- In the case of a large aircraft where gauge considerations do not apply q may well be nearly zero. On the other hand q can be of the order of 30 if the wing is structurally inefficient. Table 1 is guide to the choice of a value for q in a particular case.

The second term in Eq.(4) is the correction for practical and layout considerations. The third is the flap, aileron and shroud weight and the last the basic rib weight.

One comprehensive wing weight prediction method is that due to Burt Reference 34. Although certain aspects of this method may need updating the process is defined in such a way that this can be done readily.

B3. Fuselage structure weight

Prediction of aircraft fuselage weight is fraught with difficulties. Although the fuselage accounts for approximately as much weight as the wing much less work has been carried out in this field. The reason for the difficulties is readily apparent. Fuselages can be designed by a large number of complex loading cases and the structure is inevitably influenced by many and various detail layout considerations. Any attempt to allow for all these factors in a formula is bound to be a failure and so the only

alternatives are the use of relatively simple empirical relationships or relatively complicated prediction methods.

It is reasonable to assume that fuselage weight is related to surface area and also to the design diving speed. Evaluation of surface area in the initial project phase may not be straightforward, but it is logical that it can be defined in terms of the overall dimensions. For transport aircraft fuselages based on circular arc cross sections:-

$$S_F = 2.56LD \quad \dots (5)$$

where L is the overall length of the fuselage

D is the average of maximum depth H and breadth B of the fuselage, i.e. $D = (B+H)/2$.

This relationship also holds reasonably well for many other aircraft types but may overestimate surface area for a well streamlined, circular shape and underestimate it for a bluff, slab sided shape.

Analysis of a large number of recent aircraft indicates the following relationships for fuselage weight:-

$$W_F = c_2 [2LD V_D^{0.5}]^{1.5} = c_2 [L(B+H)V_D^{0.5}]^{1.5} \dots (6)$$

$c_2 = 0.001$ for short range transport and bomber aircraft having engines on the wings

$c_2 = 0.00085$ for long range transports

$c_2 = 0.0013$ for freighter aircraft with large pressurised loading doors and heavy floors. This also applies to light twin and executive aircraft ($2LDV^{1/2} < 2 \times 10^4$) There is a penalty in the case of transport aircraft with engines mounted on the fuselage. In this case c_2 should be factored by 1.1.

$c_2 = 0.0015$ for combat aircraft with engines in fuselage

$c_2 = 0.0016$ for naval aircraft

$c_2 = 0.002$ for single engine light aircraft.

The values of c_2 have been derived on the assumption of circular arc cross sections and should be factored to allow for slab sided cross sections, up to a maximum of 1.45 for a rectangular shape.

Alternatively for transport aircraft

$$\begin{aligned} W_F &= 0.0013W^{1.32} \\ &= 0.0125W^{1.18} \end{aligned} \left. \begin{array}{l} \text{for long range transports} \\ \text{for short range transports} \end{array} \right\} \dots (7)$$

Allowance for engines mounted on the fuselage requires the same factor as above.

A useful fuselage weight prediction method is that due to Burt and Phillips Reference 37.

This is based on an analysis of empirical information and relies upon correcting a predicted shell weight for cut outs, floors, bulkheads, etc. No consideration is given to the mounting of powerplants or main undercarriages on transport aircraft fuselages and lack of recent data implies inadequate allowance for pressurisation effects. In an attempt to improve this situation Simpson Reference 38. has undertaken a new analysis. This is based on a more theoretical approach although inevitably relies upon empirical corrections. At present it is not applicable to initial project work as it requires a knowledge of the fuselage loading, and it only applies to transport aircraft. It is anticipated that further work will remove these restrictions.

In the circumstances a more detailed consideration of the method of Burt and Phillips is in order, together with suggested modifications in the light of recent data.

a) Skin weight is given as:-

$$W_{SI} = 0.00575 S_F^{1.07} V_D^{0.743} k_1 \quad \text{lbs} \quad \dots (8a)$$

which is equivalent to a mean skin thickness of

$$4k_1 S_F^{0.07} V_D^{0.743} \times 10^{-4} \quad \text{ins}$$

k_1 is a function of the fuselage length/diameter ratio and is defined approximately as:-

$$k_1 = 0.22 + 0.36 \left(\frac{L_T}{B+H} \right) - 0.14 \left(\frac{L_T}{B+H} - 2 \right)^{1.5} \dots (8b)$$

L_T is the tail arm (ft)
The last term is zero for $\left(\frac{L_T}{B+H}\right) < 2$.

Recent data suggests that if the aircraft is a transport type with rear mounted engines:-

$$k_1 = 0.12 + 0.47\left(\frac{L_T}{B+H}\right) \dots (8c)$$

On wider body transports the skin thickness is determined by pressure considerations, and in this case the skin thickness is approximately:-

$$5pD \times 10^{-4} \text{ ins}$$

where p is the working differential pressure, p.s.i.

Thus approximately pressure dominates if:-

$$5pD > 4S_F^{0.07} k_1 V^{0.743}$$

and when this is so:-

$$w_{S1} = 0.0072S_F pD \text{ lbs} \dots (8d)$$

b) Stringer weight is given as:-

$$w_{S2} = 0.000657S_F^{1.45} V_D^{0.39} N^{0.316} k_1 \text{ lbs} \dots (8e)$$

In practice it may be possible to reduce the stringer weight when the skin thickness is determined by pressure considerations, but as there is, as yet, little evidence on the magnitude of this it is suggested that Eq.(8e) is used in all cases.

c) The weight of the standard frames required to complete the gross shell is given in terms of the sum of the skin and stringer weights:-

$$w_F = k_3(w_{S1} + w_{S2})^{1.07} \text{ lbs} \dots (8f)$$

where w_{S1} and w_{S2} are derived from Eqs.(8a) and (8e) in all cases regardless of whether the actual skin weight is derived from Eq.(8d) or not.

$k_3 = 0.11$ except for freighter types when it is 0.17.

d) The total gross shell weight is thus:-

$$W_{SG} = W_{S1} + W_{S2} + W_f \quad \text{lbs} \quad \dots (9a)$$

Eqs.(8a) or (8d); (8e) and (8f) being used.

This value of W_{SG} must be corrected for the reduction of weight due to cutouts, to give a net shell weight, W_{SN} . This is done by calculating the individual weight/unit area of the skin stringers and frames and evaluating from these the equivalent reduction in weight for each cutout.

$$W_{SN} = W_{SG} - \frac{[A_1 W_{S1} + A_2 W_{S2} + A_3 W_f]}{S_F} \quad \text{lbs} \quad \dots (9b)$$

where A_1, A_2, A_3 , are the equivalent areas of skin, stringers and frame respectively which are removed.

To the net shell weight must be added the penalties required to reinforce the cutouts and provide in-filling. This results in the modified shell weight, W_{SM} . A table of values for this is given by Burt and Phillips, including the following:-

i) Windscreens and canopies

Windscreens on large, pressurised aircraft;

$$\text{Weight} = (7.9 + 0.011V_D)A_W \quad \text{lbs}$$

$$\text{Other windscreens, weight} = (3.3 + 0.011V_D)A_W \quad \text{lbs}$$

$$\text{Fixed small aircraft canopy weight} = 0.0045V_D \cdot A_W \quad \text{lbs}$$

$$\text{Sliding small aircraft canopy weight} = 0.009V_D \cdot A_W \quad \text{lbs}$$

... (9c)

where A_W is the equivalent 'wetted' area of the windscreen and canopy, sq ft.

ii) Bomb doors

$$\begin{aligned} \text{Door weight} &= 0.76(1 + 0.0114V_D)A_W^{0.9} \quad \text{lbs} \\ \text{Door surround weight} &= \frac{(W_{S1} + W_{S2})}{2} (\alpha\beta + 5\alpha + \beta) \quad \text{lbs} \end{aligned} \quad (9d)$$

where β is the length of the cutout and α is the distance round the cross section, $\alpha\beta = A_W$ in this case.

Using recent information analysed by Simpson:-

iii) Other doors

$$\left. \begin{aligned} \text{Undercarriage, passenger and freight doors and} \\ \text{mechanism weight} &= 7.3A_W - 6 \text{ lbs} \\ \text{Freight door surround weight} &= 4.65A_W + 68 \text{ lbs} \\ \text{Other door surround weight} &= 3.64A_W + 16 \text{ lbs} \end{aligned} \right\} (9c)$$

iv) Windows

$$\text{Window and surround weight} = 8.4 N_W \text{ lbs} \quad \dots (9f)$$

where N_W is the number of windows.

e) The total weight of the fuselage is obtained by adding to W_{SM} , the modified shell weight allowances for the other structural components in the fuselage. Simplified forms of values derived by Simpson are:-

i) Floors

$$\text{Passenger floor weight} = l_c [1.26N_b + 1.31(0.66 + 0.025B)] \text{ lbs} \quad \dots (10a)$$

where l_c is the cabin length, ft, and N_b is the usual number of seats across the cabin.

$$\text{Flight deck floor weight} = l_f [2N_c + 0.91B] \text{ lbs} \quad \dots (10b)$$

where l_f is the flight deck length, ft and N_c is the number of crew seats.

$$\begin{aligned} \text{Weight of freight bay floors and structure} \\ = k_4 l_b (12 + 0.065B) \text{ lbs} \quad \dots (10c) \end{aligned}$$

where l_b is the total length of the freight bays and k_4 is the number of rows of containers which can be carried.

In the case of pure freighter aircraft the floor structure weight is about twice that given by Eq.(10a) if it is supported directly on the frames, otherwise it is up to three times this value.

ii) Pressure bulkhead

$$\begin{aligned} \text{Weight} &= 0.285pA_B + 21 \text{ lbs} \quad \dots (10d) \\ \text{where } A_B &\text{ is the bulkhead area, sq ft.} \end{aligned}$$

iii) Lifting surface attachments:-

Wing attachment weight = $0.0009NW + 45 \text{ lbs}$
 Burt and Phillips give the weight of a
 conventional tail unit attachment as $0.00022W^{1.2} \text{ lbs}$
 ... (10e)

iv) Undercarriage attachments:-

Simpson suggests for nose undercarriage, the weight
 $= 0.00282W + 7 \text{ lbs}$
 From this it can be deduced that the main
 undercarriage attachment weight = $\frac{0.0143mW}{\bar{M}}$
 ... (10f)

where m is the number of main undercarriage units
 mounted on the fuselage out of a total of \bar{M} main
 undercarriage units.

v) Engine attachment penalty

In the case of transport aircraft, particularly
 those with podded powerplants it is suggested that
 an allowance should be made for attachment weight
 $= 0.04W_p$... (10g)

where W_p is the total installed weight of
 fuselage mounted powerplants.

B4. Tail unit structure weight

Typically the tail unit weighs some 2 per cent of the
 gross weight and the most significant aspect is the effect
 on centre of gravity position. Theoretical evaluation of
 the individual weights of the fin and tailplane is not easy
 due to the difficulty of establishing design loads and the
 variety of structural and control layouts. As could be
 anticipated the most significant parameters appear to be the
 area of the tail surfaces and the design diving speed.

In the case of conventional tail unit layouts, i.e.
 those with the tailplane mounted off the fuselage and a
 single fin,

$$W_{TU} = 2.57V_D S_{TU}^{1.2} \times 10^{-3} \text{ lbs} \quad \dots (11)$$

where S_{TU} is the gross area of the tail surfaces, sq ft.

Twin fin tail units are not used frequently in new designs but such recent evidence as there is suggests that Eq.(11) applies to this configuration as well.

When the tailplane is mounted at the top of a single fin better correlation is obtained by using the tailplane area rather than the total tail surface area as the parameter. This appears to be a result of the difficulty of defining the gross fin area consistently and the relatively greater contribution this can make in high tail configurations. Thus for this configuration:-

$$W_{TU} = 5.75V_D(2S_{TAIL})^{1.12} \times 10^{-3} \text{ lbs} \quad \dots (12)$$

where S_{TAIL} is the gross tailplane area, sq ft.

In the case of tailless designs the definition of gross fin area again leads to difficulties but the data suggests:-

$$W_{FIN} = 5.75V_D S_{FIN}^{1.12} \times 10^{-3} \text{ lbs} \quad \dots (13)$$

where S_{FIN} is the gross fin area, sq ft.

There is no evidence on which to base conclusions on the weight of canard surfaces, but it seems reasonable to deduce from Eqs.(12) and (13) that Eq.(12) would apply to the foreplane surface, replacing $(2S_{TAIL})$ by (S_{FORE}) and Eq.(13) to the isolated fin, S_{FORE} being the gross foreplane area, sq ft.

Figures for recent transport aircraft suggest:-

$$W_{TU} = 0.162W^{0.83} \quad \dots (14a)$$

for conventional tail units

$$\text{or } W_{TU} = 0.186W^{0.83} \quad \dots (14b)$$

for high mounted tailplane configurations.

B5. Undercarriage

The undercarriage of an aircraft weighs about 4 per cent of the all up weight but detail layout and design can give rise to considerable variations on this figure. Apart from loading the length of the units is the most important consideration. Combat aircraft tend to be designed to

operate at relatively higher loads than transport types.
Average civil aircraft values are:-

$$W_{UC} = C_3 W \text{ lbs} \quad \dots (15)$$

where $C_3 = 0.038$ for transport aircraft

$C_3 = 0.048$ for light aircraft, below
10000 lb all up weight, approx.

Comparable combat aircraft values are:-

$$W_{UC} = C_4 W^{0.93} \text{ lbs} \quad \dots (16)$$

where $C_4 = 0.081$ for fighters and bombers

$C_4 = 0.096$ for naval aircraft

More accurate evaluation can be undertaken by a method
such as that given by Phillips Reference 37

This technique uses load and length data, the
predictions being made from graphical information. Where
possible wheel and tyre weight should be based on actual data
although this is not always convenient in the project stage.

Very approximately the static load capacity of a tyre
of typical proportions is:-

$$P_S = 4.2 \left(\frac{p}{100} \right)^{0.3} d^{2.26} \text{ lbs} \quad \dots (17a)$$

where p is the inflation pressure, p.s.i.

d is the diameter, ins.

The width/diameter ratio assumed is

$$0.285 \left(\frac{p}{100} \right)^{-0.23} \quad \dots (17b)$$

At a given pressure the load capacity is approximately proportional to the product of the diameter and width and dynamic load capacity is slightly more than three times the static value.

The weight of the tyre and tube corresponding to Eqs.(17a) and (17b) is:-

$$W_t = 0.0033 \left(\frac{p}{100} \right)^{0.62} d^{2.87} \dots (18a)$$

$$\text{or } W_t = 0.79 \left(\frac{p}{100} \right)^{0.32} d^{0.61} \times 10^{-3} p_s \dots (18b)$$

The weight of the brake material is readily evaluated from a knowledge of the forward kinetic energy which is dissipated during landing, that is brake material weight per wheel is:-

$$W_b = 0.53 f \frac{W_L}{N_W} V_A^2 \times 10^{-6} \text{ lbs} \dots (19)$$

where it is assumed that 12 lb of brake material is required to absorb 10^6 lb ft of energy.

V_A is the approach speed, knots

N_W is the number of braked wheels

f is a factor which varies between 0.7 and 1.0 and depends upon the energy dissipated by aerodynamic drag.

Total wheel unit weight is given by:-

$$\left. \begin{array}{ll} 1.43(W_t + W_b) & \text{for braked wheels} \\ 1.6W_t & \text{for unbraked wheels} \end{array} \right\} \dots (20)$$

The curve for undercarriage structure weight given by Phillips is approximately equivalent to:-

$$W_s = 0.0152 l R_n^{0.78} \text{ per leg} \dots (21)$$

where l is the distance parallel to the leg from the extended shock absorber ground contact point to attachment to airframe, ft.

R_n is the resultant factored load normal to the undercarriage leg, lbs. Thus is the leg is raked back

by an angle, θ , in the two point landing case with maximum drag and side load:-

$$R_n = \frac{\lambda W_L}{M} (\sin\theta + 0.47\cos\theta) \text{ for a main unit.}$$

where λ is the equivalent factored reaction factor, (i.e. $1.5\lambda_{\text{proof}}$ or λ_{ult} as appropriate).

The total weight of the undercarriage is obtained by adding the individual components and factoring by 1.02 for fixed or 1.07 for retracting units. A further 0.003W should be allowed for bogies when these are used.

B6. Structure associated with the powerplants

It is normal to include powerplant structure such as fixed nacelles and pylons in the main structure weight. Removable panels, doors, cowls, etc are conventionally regarded as part of the powerplant weight, but care must be taken in analysing data.

When the nacelle is largely fixed fairing, the powerplant being mounted directly to main structure a typical weight is 0.014W to 0.02W or 2 to 3 lbs/sq ft of actual surface. Larger nacelles which include the engine mounting are heavier, typically 0.03W or 4 lbs/sq ft of surface area.

In the case of podded powerplants mounted on the wing the pylon weight is approximately 0.01W or 0.17 of the powerplant weight. Pods attached to the side of a fuselage are usually attached more directly to the main structure and fixed mounting weight is less, possibly being about half of the above figures.

B7. Tail Booms

Whilst tail booms are structurally efficient the need to minimise wetted area implies the use of small cross section dimensions. Because of this they are relatively heavy. Experience suggests that the total weight of a fuselage and tail booms will not be much greater than that of a conventional fuselage designed for the same role. A weight of 5 to 6 lb/sq ft of surface is typical for booms or 0.025W to 0.03W.

B8. Powerplant weight

Where possible powerplant weight should be derived from data supplied by engine manufacturers whether this be for a design study or existing powerplant. It is sometimes helpful to use information of a more general nature, especially when initial work is being undertaken, and the following summarised trends are intended to be a guide in these cases.

B8.1 Piston engines

Piston engine applications are now limited to relatively small light aircraft where the cost of a turbine power unit cannot be accepted. The majority of the engines used are air cooled and employ a horizontally opposed, in line cylinder arrangement. In this case the basic powerplant power to weight ratio is:-

$$\begin{aligned} \left(\frac{\text{HP}}{\text{W}}\right) &= 0.34(1 + 0.0043\text{HP}) \text{ for } 0 \leq \text{HP} \leq 200 \\ &= 0.65 \text{ for HP} > 200 \text{ unsupercharged} \\ &= 0.6 \text{ for HP} > 200 \text{ supercharged} \\ &= 0.9 \text{ for unsupercharged Continental 'Tiara'} \\ &\quad \text{type engines of about 400 HP.} \end{aligned}$$

B8.2 Propellers

The weight of a propeller varies according to size and the type of pitch control used.

Large variable pitch propellers of recent design weigh about 0.22 lb/HP capacity inclusive of control gear. Small fixed pitch propellers have a similar weight characteristic but variable pitch propellers for small piston engine applications weigh from 0.25 to 0.3 lb/HP.

B8.3 Turbine engines

A survey of recent turbojet and torbofan engines suggests the following relationship for engines designed to the current state of the art:-

$$\left(\frac{T}{W}\right)_e = K_1 \left[5 + \frac{R}{10} - \left(1 - \frac{T}{10000}\right) \right] \text{ lbs} \quad \dots (22a)$$

where the last term is zero for $T \geq 10000$ lbs.

R is the static bypass ratio

K_1 is a coefficient which is unity for propulsion engines with bypass ratios of up to about 8 and 3 for pure lift engines. When bypass ratios of the order of 10 or more are used for propulsion engines it is necessary to use a geared fan, possibly with variable pitch blades and in this case K_1 is 0.75 approximately. It is in excess of unity, possibly around 1.4 for vectored lift engines.

Eq. 22a includes allowance for engine accessories and applies to units with afterburning as well as without. Afterburner weight is included.

Suitable adaptation of Eq. 22a enables the weight of turboprop engines to be estimated. If propeller weight is included and a typical static thrust is taken as 4 lb/HP with a static bypass ratio of 50:-

$$\left(\frac{HP}{W}\right)_{e+p} = 0.175 \left[10 - \left(1 - \frac{HP}{2500}\right) \right] \text{ lbs} \quad \dots (22b)$$

where the last term is zero for $HP \geq 2500$ and K_1 has been taken as 0.75.

When allowance is made for propeller weight at 0.22/HP Eq. 22b reduces to 2.85 for engines in excess of 2500 HP.

B8.4 Powerplant installation

It is convenient to allow for the weight of the installation of a powerplant by appropriately factoring the basic engine weight. This can lead to difficulties when comparing data especially as air intakes for buried turbojet installations may be included in the structure weight. The suggested factors are:-

Small single air cooled engine installations	1.3
Small twin air cooled installations	1.4
Larger multi air cooled engine installations	1.6
Liquid cooled engine installations	1.9
Turboprop installations	1.7

All the above are based on engine weight exclusive of propeller, the propeller weight being in the installation factor.

Turbojet engines buried in wing	1.25
Turbojet engines buried in fuselage	1.2
Turbojet engines alongside fuselage	1.3
Podded turbojets on wings	1.05 to 1.2
depending on use of silencers and thrust reversers	
Podded turbojets on fuselages	up to 1.3
Supersonic installations, based on weight of engine inclusive of reheat, but excluding intakes	1.1 to 1.2

B9 Fuel and Oil Systems

For convenience oil systems have been considered in conjunction with the fuel system but their weights are only significant in the case of large piston engines.

B9.1 Tankage

It is normal for tank weight to be included as part of the fuel system or structure when the tanks are not removable. As a guide for purposes of comparison the following figures are quoted:-

Rigid metal tanks	$10 + 0.69V$	lbs
Rigid fibreglass tanks	$7 + 0.4V$	lbs
Crash proof tanks	$7 + 0.38V$	lbs
Flexible bag tanks	$10 + 0.12V$	lbs
Drop tanks, approximately	1.0V	lbs

where V is the tank volume in gallons.

Flexible tanks require some degree of plating to support them and at most this can about double the tank weight, but like the weight of sealing for integral tanks it is part of the structure. Sealing for integral tanks is likely to amount to about 0.1V.

B9.2 Residual fuel

There is always some fuel in the tanks which cannot be used. Obviously the quantity must vary according to the number and shape of the tanks and should be as small as possible. It is usually quoted as part of the basic weight of an aircraft separately from the fuel system weight but for project design work it is convenient to include them together. Residual fuel can amount to more than 1 per cent of the gross aircraft weight on military aircraft with complex tank systems although 0.5 per cent is more typical. In the case of civil aircraft with integral tanks a more usual figure is 0.25 per cent.

B9.3 Fuel system

As might be expected considerable variations are observed in the weight of fuel systems. For combat aircraft with integral fuel tanks the weight is likely to be between $0.017W$ and $0.027W$ where W is the gross aircraft weight. Values in excess of $0.04W$ are possible if the system is complex and uses separate rigid or flexible tanks.

There is a definite trend in the case of civil aircraft with integral tanks, the system weight being on average:-

$$W_{FS} = 0.03W^{0.8} \quad \dots (23)$$

This is likely to be approximately doubled if separate tankage is used. Fuel systems on supersonic transports are more complex than on subsonic ones and a factor of about 1.4 is required on the values given by Eq.23

Residual fuel weight is included in all the above figures.

B10 Power Services, Including Flying Controls

Correlation of information on the hydraulic, pneumatic electrical and flying control systems is very difficult. Necessarily there will be large variations according to the role of an aircraft and the way in which it is operated and in most instances it is possible only to give general trends.

B10.1 Flying control systems

There is a definite tendency for the percentage weight of flying control systems to fall with increase in aircraft size although naturally the use of powered systems introduces a step into the general picture.

The evidence available suggests that on average the weight of a manually operated flying control system is given by:-

$$W_{FC} = 0.16W^{0.75} \quad \dots (24a)$$

except when the aircraft is a two seat trainer with dual control when

$$W_{FC} = 0.23W^{0.75} \quad \dots (24b)$$

Whilst considerable variation can be expected in practice Eqs. 24a and 24b should provide an adequate guide for initial design work.

Powered flying control systems generally fall between the above two figures and it is suggested that in this case:-

$$W_{FC} = 0.2W^{0.75} \quad \dots (24c)$$

One difficulty in interpreting this value in relation to existing aircraft is the ill defined division between the weight of the flying controls and the appropriate power service.

When a supersonic aircraft incorporates variable air intakes an additional allowance of $0.005W$ should be made.

B10.2 Hydraulics and pneumatics

The hydraulic and pneumatic systems are associated together in that in the great majority of cases they provide the power for most of the major actuation requirements. Considerable variation in the weight of these systems is likely depending upon the number of individual services which have to be provided and the degree of multiplication employed. As might be expected one of the dominant parameters is the use or otherwise of powered controls.

Typical cases when power controls are not fitted suggest the relationship:-

$$W_H = 0.155W^{0.74} \quad \dots (25a)$$

It is advisable to increase this value if the system is particularly complex and vice-versa.

The equivalent equation for aircraft with powered controls is:-

$$W_H = 4.8W^{0.5} \quad \dots (25b)$$

This gives good correlation for civil transport aircraft if the weight of the flying controls is considered with it

B10.3 Electrics

Although a great deal of variation is inevitable the following relationships are a good guide as to what allowance should be made for initial design work:-

Civil transport aircraft of all types:-

$$W_E = 1.2W^{0.67} \quad \dots (26'a)$$

Light aircraft not used in public transport category:-

$$W_E = 0.0028W^{1.26} \quad \dots (26b)$$

Supersonic strike/bomber aircraft:-

$$W_E = 0.1W^{0.84} \quad \dots (26c)$$

Other military aircraft:-

$$W_E = 0.18W^{0.84} \quad \dots (26d)$$

In some cases the quoted weights of the electrical installation and the electronic equipment are ill defined and can be misleading.

B10.4 Accessory drives and auxiliary power unit

On some aircraft the accessories such as alternators and pumps are mounted away from the powerplant on an auxiliary gearbox. When this is the case an allowance of about 0.005W is necessary to cover the weight of the gearbox and remote drives.

Auxiliary power units vary in installed weight from approximately 100 lb for a small unit suitable for use on a light transport to 800 lb on a large transport. A minimum allowance of about 400 lb is required for a first line airliner.

B11. Equipment

The equipment is taken to include instruments, radio, radar, automatic controls, navigation, safety protection such as fire precautions and de icing and any special handling equipment. External paint may also be covered under this heading.

B11.1 Instruments and automatic controls

The weight of the basic flight and engine instruments on a light aircraft is 30 lbs to 40 lbs and this rises to about 65 lbs on somewhat more sophisticated designs. Transport aircraft intended for international operations carry about 500 lbs of instruments unless extensive auto controls are incorporated with multiplexing when it may be as much as 1500 lbs. The minimum allowance for internal transport operations appears to be in the region of 100 lbs to 150 lbs.

B11.2 Radio, radar and navigation equipment

Although certain minimums can be established the weight of the electronic equipment in civil aircraft is largely determined by the requirements of the operator in relation to the routes on which the aircraft is to be used. Whilst a simple communications radio installation accounts for only

about 25 lbs an allowance of about 100 lbs appears to be necessary to meet the minimum requirements for a light transport type operated internally. More general international operation requires an allowance of 350 lbs to 400 lbs whilst in the case of airliners intended for world wide operation 2000 lbs is an approximate allowance. This can rise to more than 3000 lbs if considerable duplication of systems is necessary.

Equipment installed in military aircraft is often effectively part of the payload and hence may vary enormously. Typically it accounts for 0.025W to 0.045W, with the lower figures for high performance supersonic aircraft. In an actual case the equipment, which usually includes instruments, is specified according to the role.

B11.3 De-icing

De-icing is not always fitted and when it is the coverage can vary considerably. In the case of transport aircraft and longer range bombers it is likely that icing conditions will be met relatively frequently. The same is true of naval aircraft.

For bombers and transport aircraft a typical allowance is:-

$$W_D = 0.2W^{0.7} \quad \dots (27a)$$

and it may be twice this for some naval types.

In the case of military strike aircraft:-

$$W_D = 0.09W^{0.7} \quad \dots (27b)$$

When an aircraft is designed to fly supersonically for long periods it is not normally necessary to provide for full airframe de-icing although engine air intake protection is essential. The allowance for this is approximately:-

$$W_D = 0.016W^{0.7} \quad \dots (27c)$$

B11.4 Fire precautions and tank protection

There appears to be little logical explanation for the relatively large variations in quoted weights of fire and tank protection. Average values for use as a design allowance are $0.006W$ for military and $0.003W$ for civil aircraft.

B11.5 External paint

The majority of aircraft have an external coat of paint for decorative, camouflage and protective purposes. An approximate allowance for this is $0.1S$ lbs where S is the wing area in sq ft.

B12 Furnishings

As well as obvious items such as seats and internal decor the furnishings includes the airconditioning and pressurisation systems, removable partitions, sound insulation, galley, toilets, and all other items associated with comfort of the crew and passengers.

B12.1 Seats

Tourist class seats in airliners weigh about 20 lb per passenger although for short range operations seat weight can be as low as 17 lb per passenger. The average seat weight for mixed class flights is likely to be nearer 25 lb per passenger.

The seat and harness allowance on light aircraft is 24 lbs to 30 lbs. Ejector seats weigh about 130 lbs.

B12.2 Furnishings

For initial design work it is convenient to consider all the furnishings, including seats but excluding air conditioning, as a single item. Individual operators fit furnishings to suit their own requirements.

The total furnishing allowance for airliners is of the order of 100 lbs per passenger, although it can be as low as 40 lbs per passenger for a small feeder line aircraft. In the case of long range transports there is reasonable consistency of total furnishing weight as:-

$$W_F = 0.004W^{1.2} \quad \dots (28a)$$

whilst for short range transport the comparable figure is:-

$$W_F = 0.0075W^{1.2} \quad \dots (28b)$$

Furnishings in light aircraft often include little more than the seats.

In the case of military aircraft the total furnishing weight may be taken as approximately 50lbs per member of crew unless ejector seats are fitted when 200 lbs per member of crew is more typical.

B12.3 Air conditioning, pressurisation and oxygen

The weight of the air conditioning and associated systems could well be expected to be related to the cabin size or number of people carried by the aircraft. Correlation on these bases is made difficult by the differing operating requirements and substantial hardware developments in recent years. With current technology and comfort standards the air conditioning in a pressurised civil aircraft has a gross weight of about 18 lb per passenger. Correlation on an aircraft weight basis suggests:-

$$W_{AC} = 0.048W^{0.88} \quad \dots (29a)$$

For unpressurised light transports:-

$$W_{AC} = 0.024W^{0.88} \quad \dots (29b)$$

and a typical relationship for military aircraft is:-

$$W_{AC} = 0.009W \quad \dots (29c)$$

B13 Total Weight of Power Services, Systems, Equipment and Furnishings

It is often noticed that whilst there may be considerable variation in the weight of the individual components discussed above, the total weight of them shows much less variation. Analysis of the total weights covered in paragraphs B9 to B12 inclusive for over 40 civil aircraft of all types suggests the following typical values for the difference between the total empty weight and that of the sum of the structure and powerplant:-

- a) Light aircraft of up to 10,000 lb gross weight
0.05W at 1500 lb increasing linearly
to 0.17W at 10,000 lb.
- b) 10,000 lb < W < 250,000 lb.
0.17W, but varying between 0.15W and 0.2W
with up to 0.25W for executive aircraft.
- c) Large transports, long range W > 250,000 lbs.
0.17W at 250,000 lbs falling linearly to a
minimum value of about 0.1W above 400,000 lbs.

TABLE B1

WING WEIGHT PREDICTION

Typical Values of Design Factor, q.

Aircraft Type	Range of q	Mean Value of q^*
Long range transports	$-6 < q < 4$	0
Short/medium range transports	$10 < q < 25$	18
Light, executive trainers	$4 < q < 10$	8
Combat aircraft	Wide scatter	20

* Based on use of stated relief factors, $\sum \Delta r$

APPENDIX C

Noise Prediction and Comparison

(This appendix is based on Lecture Notes DES 7309 and 7411 prepared by the author)

C1. POWERPLANT NOISE

It is possible to divide the noise generated by a turbine powerplant into three main components, the fan and compressor, the turbine and the jet effects. The changes in engine technology in recent years have been partly determined by the need to reduce the powerplant noise level and an important effect of the developments has been a redistribution of the major noise sources. Early turbojet engine noise was due largely to jet effects whilst with higher bypass ratio engines of more recent design the compressor noise has become more dominant. This is illustrated by Figure C1 which has been extracted from a paper by R.H. Weir (Reference 38). The diagram also gives an indication of the directional nature of the noise.

C1.1 Compressor and fan noise

Increase of engine bypass ratio has the effect of increasing the noise from the fan and compressor assembly.

Some early jet engines using centrifugal compressors in which case the noise is due to a combination of turbulence, and unsteady flow, both of which are force effects with a theoretical (velocity)⁶ characteristic. Axial compressors are more complex. Turbulence and boundary layer pressure effects are present both of which are force effects. However there are also wake effects which are a type of flow noise. Noise from axial compressors has been discussed by a number of authors, for example Bragg and Bridge (Reference 39). They suggest that the noise is (velocity)^{5.8} dependent and quote formulae for evaluating the noise output:-

(a) Installed compressor:-

$$(PWL^* - 10 \log M) = 128 + 47.6 \log \left[\frac{V_{TIP}}{1000} \right] \quad \begin{matrix} \text{dB/unit} \\ \text{mass flow} \end{matrix}$$

... (1a)

(b) Stalled compressor:-

$$(PWL* - 10 \log M) = 132 + 34.4 \log \left(\frac{V_{TIP}}{1000} \right) \text{ dB/unit mass flow} \quad \dots (1b)$$

where M is the flight Mach No.

V_{TIP} is the compressor tip speed in ft/sec.

The mass flow is in lb/sec units.

$$(PWL* = (PWL)_{\text{measured}} - 10 \log(1 - M_n)$$

where $(PWL)_{\text{measured}}$ is the actual measured sound power level and M_n is the component of the Mach No. through the compressor rotor in the direction of the noise.

The sound pressure level, SPL, at 500 ft from the source is given approximately by

$$SPL = PWL - 54 \quad \text{dB} \quad \dots (2)$$

These formulae were derived from tests on engines of low bypass ratio, that is between 0 and 1.

Engine fan noise is dependent upon detail design features and can be attenuated significantly by appropriate treatment of the duct. One important development was the elimination of the inlet guide vanes. A large part of fan noise arises from the interaction between the distortion pattern of the air entering the inlet and fan, or emerging from the fan and outlet guide vanes. These effects produce virtually all the noise when the relative rotor tip speed is subsonic. At supersonic speed however, there is a large contribution from so called 'buzz-saw' effect which is a shock wave effect around the circumference of the fan.

One paper which discusses the prediction of fan noise is that by Sutcliffe, Merrick and Howell, Reference 40.

The noise level suggested for both fans and compressors is given by two formulae:-

$$PWL = 103.3 + 17.7(HP) - 2.7 \log(N) \quad \text{dB} \dots (3a)$$

$$\text{or } PWL = 106.9 + 10 \log(HP) - 10 \log(\Delta T) \quad \text{dB} \dots (3b)$$

where HP is the compressor or fan HP

N is the number of rotor blades

ΔT is the stage temperature rise.

The first of these two equations is more applicable to the compressor whilst the second is a revised version which was specifically intended to apply to fans.

The values given for PWL in Eqs.(3) can be converted to perceived noise values by using eq.(2) and adding a subjectivity allowance,

Although quoted as 14 dB it is often taken as 12.5 dB for compressors.

Some information on fan/compressor noise as a function of bypass ratio has been given by West, Reference 41,

primarily in connection with VTOL engines of the RB202 type. For a total datum thrust of 80,000 lbs the perceived noise at 500 ft radius is approximately given as:-

With inlet guide vanes:-

$$PN_{dB} = 108 + 24 \log R \quad (4a)$$

Without inlet guide vanes:-

$$PN_{dB} = 93 + 10 \log R \quad (4b)$$

where R is the bypass ratio.

Similar trends are quoted by Armstrong and Jones Reference 42, in respect of RB211 type engines. Silencing treatment in the fan duct reduces the above noise levels by effectively 5 to 7 dB. Thus current trends suggest that new fan designs should have noise levels of approximately:-

$$PN_{dB} = 88 + 10 \log R + 10 \log\left(\frac{T}{80000}\right) \dots (4c)$$

at 500 ft

where T is the engine static thrust in lbs.

C1.2 Turbo-machinery - Core engine

Although the turbo machinery noise by itself may well be less than the fan or jet noise it is directional and does interact with the jet noise. One formula which has been used and can also be applied to compressors is that given by Smith and House, Reference 43.

$$\text{SPL} = 50 \log\left(\frac{V_{\text{rel}}}{1000}\right) + 10 \log m + \alpha + \Delta F + 75 \text{ dB} \dots (5)$$

at 500 ft

where V_{rel} is the relative blade tip speed (ft/sec)

m is the mass flow

α is the mean incidence deviation from the peak of the lift curve slope (degrees)

ΔF is a flow correction factor to apportion energy in the front and rear arcs

The relative importance of the turbine noise has been discussed by Dawson and Sills, Reference 44.

It is shown that the contribution from the turbine becomes very important as the jet velocity decreases, especially when it is less than sonic. The turbine noise level suggested by West for the 80,000 lbs equivalent thrust VTOL engines at 500 ft is approximately:-

$$\text{PNdB} = 115 - 17 \log R \dots (6)$$

This can be reduced by placing noise attenuating linings in the ducting.

C1.3 Jet Noise

Dawson and Sills give data on combined internal turbine and tailpipe noise and external jet noise. The quoted external jet noise from the model test results, measured at 105° in azimuth from the intake axis and corrected to an overall datum value is:-

$$\text{SPL} = 143 + 80 \log\left(\frac{V_j}{a}\right) \text{ dB} \dots (7)$$

where V_j is the jet velocity

a is the speed of sound.

The coefficient of 80 in the velocity term demonstrates the quadrupole nature of the sound.

The added effect of internal noise is complex. At higher values of (V_j/a) the total noise is given by:-

$$\text{SPL} = 150 + 56 \log(V_j/a) \text{ dB} \dots (8)$$

However for $\log(V_j/a) < -0.1$ there is an increasing increment in SPL above the values given by the relationship of Eq.(8), up to 10 dB at $\log(V_j/a) = -0.5$.

Specific evaluation of near field jet noise levels at all angular locations is given in ESDU Data Sheets Fatigue 70002. This information is primarily for evaluating jet noise effects on structures.

The variation of jet noise with bypass ratio as given by West for the 80,000 lbs VTOL design at 500 ft is approximately:-

$$\text{PNdB} = 128 - 30 \log R \quad \dots (9)$$

Some jet noise reduction is possible by varying the hot nozzle area (see Armstrong and Jones), or by increasing the circumference of the nozzle by corrugations to obtain better mixing, see Figure C2.

C1.4 Total engine noise

In compiling the total noise due to the powerplant it must be remembered that all the noise levels are given on a logarithmic scale.

(a) Shaft turbine engines.

A reference which attempts to quote a formula for evaluating total noise for shaft turbines, especially helicopter powerplants, is that by Davidson and Hargest, Ref.50. This paper gives compressor noise as approximately:-

$$\text{PWL} = 89 + 17.7 \log \text{HP} \quad \text{dB}$$

where HP is the HP of the first compressor stage.

It is suggested that 3 dB should be added to cover directional effects, turbine noise, etc. Relating the sum to SPL at 500 ft distance:-

$$\text{SPL} = 37.5 + 17.7 \log \text{HP} \quad \text{dB}$$

Allowing for subjectivity:-

$$\text{PNdB} = 50 + 17.7 \log \text{HP} \quad \dots (10)$$

which is the total noise level for a shaft turbine.

(b) Fan engines

It is necessary to add an allowance for both fan and jet noise to Eq.(10) to obtain the total noise of a vertical or propulsion engine. Using the data given by West quoted in Eq.(4), (6) and (9); the total noise of a fan engine with silencing and no inlet guide vanes RB 202 or RB 211 standard at 500 ft:-

$$\begin{aligned} \text{PNdB} = 10 \log & \left[\text{Antilog} \left\{ \frac{88+10 \log R}{10} \right\} + \text{Antilog} \left\{ \frac{115-17 \log R}{10} \right\} \right. \\ & \left. + \text{Antilog} \left\{ \frac{128-30 \log R}{10} \right\} \right] + 10 \log \left(\frac{T}{80000} \right) \dots (11) \end{aligned}$$

This is the full thrust condition and reduction is obtained by reducing thrust; approximately

$$\Delta \text{PNdB} = -15 \left(\frac{T - T_R}{T} \right) \dots (12)$$

where T_R is the reduced thrust.

Recent developments in silencing techniques should enable further reductions, probably of the order of 5 dB to be made. Note that when the bypass ratio is about 10, the noise from each of the main sources is roughly equal.

C1.5 Propeller noise

Propeller noise at maximum power is apparently due to wake noise. That is it is a function of the eighth power of the velocity of air flowing over the tips. Since the power is a function of the cube of the velocity the noise is proportional to $(\text{HP})^{8/3}$. There is also a low frequency noise component, below about 300 Hz.

Actual noise level has been given by Fleming, Reference 45 as:-

$$\text{PWL} = 31 + 27 \log \text{HP} \quad \text{dB}$$

Thus the subjective propeller noise at 500 ft is approximately:-

$$\text{PNdB} = 27 \log \text{HP} - 10 \dots (13)$$

C2. Airframe Noise

Until recently it has been considered that as far as fixed wing aircraft are concerned the only significant noise source was the powerplant system. However with the recent developments in quieter engines it has been realised that the noise produced by the airflow over the aircraft may become relatively more important. There is not a great deal of evidence available. Dawson and Sills Reference 46

report on work by Blumenthal, Streckenback and Tate, Reference 47, and Gibson, Reference 48.

Tests have been carried out on Boeing 727 and 747 aircraft from which it would appear that the noise is approximately proportional to the weight. It is to be expected that the noise will be of the dipole variety since this covers boundary layer pressure fluctuations, turbulence interaction, vortex shedding, etc. Thus, by comparison with rotor blade experience quoted in 1.1.4 it is reasonable to assume that the noise is a function of $C_L^2 S V^6$. The information given in the paper by Dawson and Sills is for the aircraft on the approach, presumably the noise being measured one mile from the threshold according to FAR Part 36. With a conventional 3 degree glide slope this implies an aircraft altitude of about 300 ft. From these very limited results it appears that in this condition the airframe noise is:-

$$PNdB = 10 \log S + 20 \log C_L + 60 \log V - 85$$

or correcting to the 500 ft distance approximately:-

$$PNdB = 10 \log S + 20 \log C_L + 60 \log V - 89 \quad \dots (14a)$$

In this form the formula does not immediately indicate the relative importance of design parameters due to the interdependence of C_L , S and V in the approach condition. If C_L is replaced the dependence is seen to be a function of $(\frac{W}{S}) \cdot W V^2$ which is a more useful form, whence Eq.(14a) becomes:-

$$PNdB = 10 \log(\frac{W}{S}) + 10 \log W + 20 \log V - 31 \quad \dots (14b)$$

where in Eq.(14) S is in sq ft, W in lbs and V in ft/sec.

Thus the airframe noise is directly proportional to the wing loading, the weight and the square of the velocity. Eq.(14) has been derived from a very small amount of data but it is worthy of comparison with the comparable formula for rotor noise, Eq.(15). The comparison might suggest that the airframe equation is somewhat optimistic.

It should be noted that when the lift is power augmented there will be a further increment to the noise. For example some NASA results quoted for a blown flap configuration by Dorsch, Kreim and Olsen, Reference 54, suggest that there is perhaps about a 5 dB penalty due to 'scrubbing' of the jet and other causes. Further results are given by Gibson, Reference 53, for internally blown, externally blown and augments wing flap arrangements. The internally blown system seems to be some 10 dB quieter than the augments wing for a given pressure ratio with the externally blown flap system the noisiest of all. However the latter is typically designed to operate at a lower pressure ratio.

C3. Rotor Noise

Because of the special noise problem associated with rotorcraft there has been a great deal of investigation into the noise of rotors. Tail rotors as well as the main rotors can have significant effects. The main causes of rotor noise have been discussed in 1.1. Leverton Reference 49, has suggested that the aim in rotor design should be to ensure that the broadband noise exceeds the rotational noise. It has been shown that rotational noise is dominant for two bladed rotors. Leverton suggests that as the number of blades is increased the rotational noise decreases by about 4-5 dB for each additional blade, other items being identical. On the other hand the broadband noise will decrease only by $10 \log (B_1/B_2)$ where B_1 and B_2 are the relative number of blades. An interesting observation about rotational noise is that in some circumstances it may follow a V^{10} law whilst broadband noise is V^6 . Leverton quotes the following tip speeds, above

which rotational noise is dominant:-

2 blades	400 ft/sec
4 blades	680 ft/sec
6 blades	720 ft/sec

C3.1 Rotor Broadband Noise

Rotor noise has also been discussed by Davidson and Hargest. Using the Yudkin dipole as modified by Goddard and Stuckey it is suggested that the broadband noise at 500 ft is given by:-

$$\text{SPL} = 60 \log V_t + 20 \log C_{L_t} + 10 \log S - 84 + f(\lambda) + f(\theta) + f(k) \quad \text{dB}$$

where C_{L_t} is the tip C_L and V_t the tip speed, S is total blade plan area in rotor in sq ft, $f(\lambda)$, $f(\theta)$, $f(k)$ are functions of airspeed, directivity of noise and wind speed respectively. Values are given in the reference for these functions. Approximately:-

$$\begin{aligned} f(\lambda) &= 14\lambda \quad \text{dB} \quad \text{for a two bladed rotor} \\ &= 8\lambda \quad \text{dB} \quad \text{for a multi blade rotor} \\ &= 0 \quad \text{for a convertible rotor which is} \\ &\quad \text{normally aligned perpendicularly to} \\ &\quad \text{the flight direction} \end{aligned}$$

λ is the advance ratio (airspeed/tip speed)

$f(\theta)$ varies from 0 to -15 dB approximately at 500 ft distance according to wind conditions and the location of the observer relative to the rotor; 0 dB for immediately below; -15 dB when alongside.

$$f(k) = 0.3V_w \quad \text{dB}$$

where V_w is the windspeed in knots.

There is also a subjectivity factor. This varies according to windspeed, rotor tip speed and blade chord. It varies from as little as 4 dB when both wind and tip speed are low and the blade chord is large to 12 dB in the opposite circumstances. Leverton suggests that the total PNdB given by Davidson and Hargest is some 2 dB too high so the following corrected approximate formula can be stated for 500 ft distance vertically above the observer.

In zero wind, low tip speed and wide chord multiple blades:-

$$\text{PNdB} = 60 \log V_t + 20 \log C_{L_t} + 10 \log S + 8\lambda - 82 \quad \dots (15a)$$

In winds above 10 knots. with high tip speed and narrow chord multiple blades:

$$\text{PNdB} = 60 \log V_t + 20 \log C_{L_t} + 10 \log S + 8\lambda + 0.3V_w - 74 \quad \dots (15b)$$

Cheeseman, Reference 72, states these formulae in an alternative form which is particularly appropriate to vertical flight where $\lambda = 0$; and for zero wind

$$\begin{aligned} \text{PNdB} &= 40 \log V_t + 10 \log C_{L_t} + 10 \log L - 52 \\ \text{and} \quad &40 \log V_t + 10 \log C_{L_t} + 10 \log L - 44 \text{ respectively} \end{aligned}$$

where L is the total thrust of the rotor and further

$$C_{L_t} = \frac{2C_t}{\sigma}$$

where C_T is the thrust coefficient and σ the solidity.

It is interesting to compare Eq(15a) with Eq(14a) for airframe noise. Considering the rotorcraft hovering case there is apparently a difference of 7 dB in favour of the fixed wing aircraft. With the allowance for a comparable forward speed this increases to about 10 dB. The significance of this is not immediately apparent although it may well be a function of the different powers required.

C3.2 Rotational Noise

The evaluation of rotational noise is complex requiring the analysis of a large number of noise harmonics. One method has been given by Lawson and Ollerhead Reference 52. The problem and its solution is also discussed in the paper by Leverton.

C3.3 Blade Slap

Blade slap has been investigated by Leverton who has proposed a blade slap factor, BSF where :-

$$BSF = V_t^2 L^2 / \bar{R}.B \quad \dots (16)$$

where \bar{R} is the rotor radius (ft) and B the number of blades. If the BSF is greater than $7 \text{ to } 9 \times 10^9$ the blade slap is likely to be unacceptable.

C4.0 Gear and Transmission Noise

Gear noise in aircraft can be important, especially in the case of rotorcraft. Whilst the most significant effect is likely to be on internal noise levels, at certain frequencies the transmission noise adds appreciably to other external noise components. The noise of gears arises from the meshing of the teeth and is thus a function of tooth shape. There have been suggestions that conformal gearing is noisier than involute gearing but whether this is so for comparable power transmission has yet to be conclusively established. The noise is a function of the square of the speed of revolution but the apparent effect of this is modified by subjectivity considerations. For further information see, for example, Grover and Anderton 2nd International Power Transmission Conference, London 1971.

C5.0 Effect of Forward Speed on Aircraft Noise

The effect of forward speed on rotorcraft broad band and rotation noise is discussed in 4 above. Airframe noise is also obviously critically dependent upon the forward speed, as in 3 above.

As far as the effect on the powerplant is concerned the situation is somewhat obscure. Some tests have suggested that rear arc noise, that is turbine and exhaust noise, decreases with increase of forward speed. Tests on engines of low bypass ratio and with inlet guide vanes suggest the opposite trend for the compressor noise component. On the other hand the RB 211 experiences a very appreciable reduction of inlet/fan noise with forward speed increase.

C6 Noise of powered lift systems (see also paragraph C2)

A number of papers have been published on the noise of powered flap systems. General deductions from these are difficult to achieve as the noise level appears to be dependent upon many parameters, including the experimental set up. However some general trends are apparent and worthy of comment.

C6.1 Externallyblown flaps

The noise is dependent upon the fan velocity, core velocity, nozzle pressure ratio, flap deflection and nozzle area and probably on other parameters also. Useful references are 54 and 55.

The evidence suggests that the subjectivity factor is 8 to 9 dB and that the sideline noise at 500 ft is some 5 dB less than the flyover noise at the same distance.

In the case of a double slotted flap system with the rear element deflected to twice the angle of the forward segment the noise level increases by some 4 dB for each $10^{\circ}/20^{\circ}$ increase of flap angle.

Nozzle velocity and pressure ratio are related. The noise penalty for increase of fan velocity varies from about 2.5 dB/100 ft/sec at 500 ft/sec to about 1.5 dB/100 ft/sec increase at 1000 ft/sec.

Scaling for nozzle area appears to be approximately linear providing the total engine exhaust area is used, in conjunction with core pressure ratio.

The data suggests that for a datum 10,000 lbs thrust engine with a fan velocity of 750 ft/sec and flaps set at $10^{\circ}/20^{\circ}$ the flyover noise level at 500 ft is about 105 PNdB. In a typical landing case with the flaps at $20^{\circ}/40^{\circ}$, but reduced thrust the flyover noise is about 107 PNdB.

One of the problems of the externally blown flap is the difficulty of reducing the noise level by acoustic treatment, etc. Some early suggestions that a correctly designed mixer nozzle would reduce the noise level by 5 dB would seem to be unjustified. It can be concluded that whilst the externally blown flap is basically simple in concept the noise problem is extremely severe.

C6.2 Augmentor wing

The subjectivity factor for the augmentor flap system is about 1 dB higher than that of the externally blown flap. In addition to the sources quoted for the latter Reference 53 is of use.

An increase of nozzle pressure ratio of unity in the range 1.4 to 2.4 results in an increase in the noise level of 15 dB.

There is little evidence on the effect of flap angle but an increase of about 4 dB is suggested for angle increase from 50° to 75° .

As with the externally blown flap the sideline noise level is apparently some 5 dB less than the flyover value.

For a given nozzle pressure ratio and velocity the noise level is approximately directly proportional to nozzle area.

For a nozzle pressure of 1.8 corresponding to a velocity of 1000 ft/sec and a datum thrust of 10,000 lb the flyover noise level with 20° flap setting is about 99 dB. The landing flyover noise will be about 102 dB, depending on flap setting.

One advantage of the augmentor wing is the possibility of silencing the system. Some investigations by Boeing are reported in Reference 29. The use of acoustic liners in the flaps ducting should result in some 8 dB reduction at about 5000 Hz. The nozzle geometry is critical. If a multi-segment nozzle is used with screech shields a further 7 dB reduction for nozzle pressure ratios of about 2 should be possible, making a total of 15 dB. This is so significant that it is likely that the basic engine noise will dominate in this case. The weight penalty for silencing is quoted to be equivalent of about $0.0014W$ per dB reduction (W being the gross aircraft weight).

C6.3 Internally blown flaps

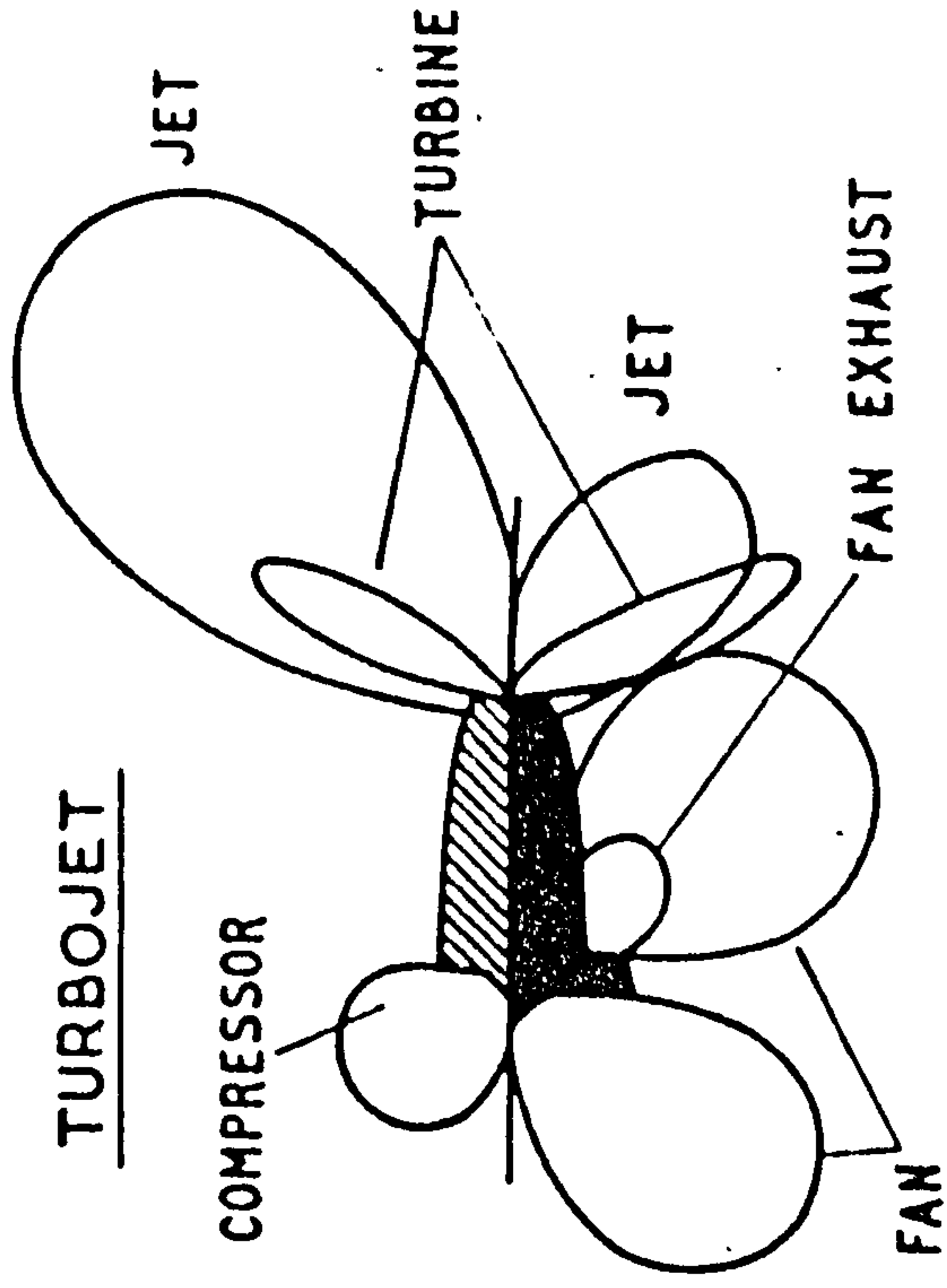
Internally blown flap and augmentor flap noise appear to follow very similar trends except that for a given nozzle pressure ratio the former is considerably quieter. The magnitude of the difference is not well established but would appear to be around 10 dB. It must be pointed out, however, that the practical nozzle pressure ratio for an internally blown flap is much higher than that used for the augmentor systems. This suggests that in general the augmentor system is likely to be quieter and it is also more amenable to quietening. The augmentor pressure ratio is more easily matched to the characteristics of high bypass ratio engines.

C6.4 Over the wing blowing

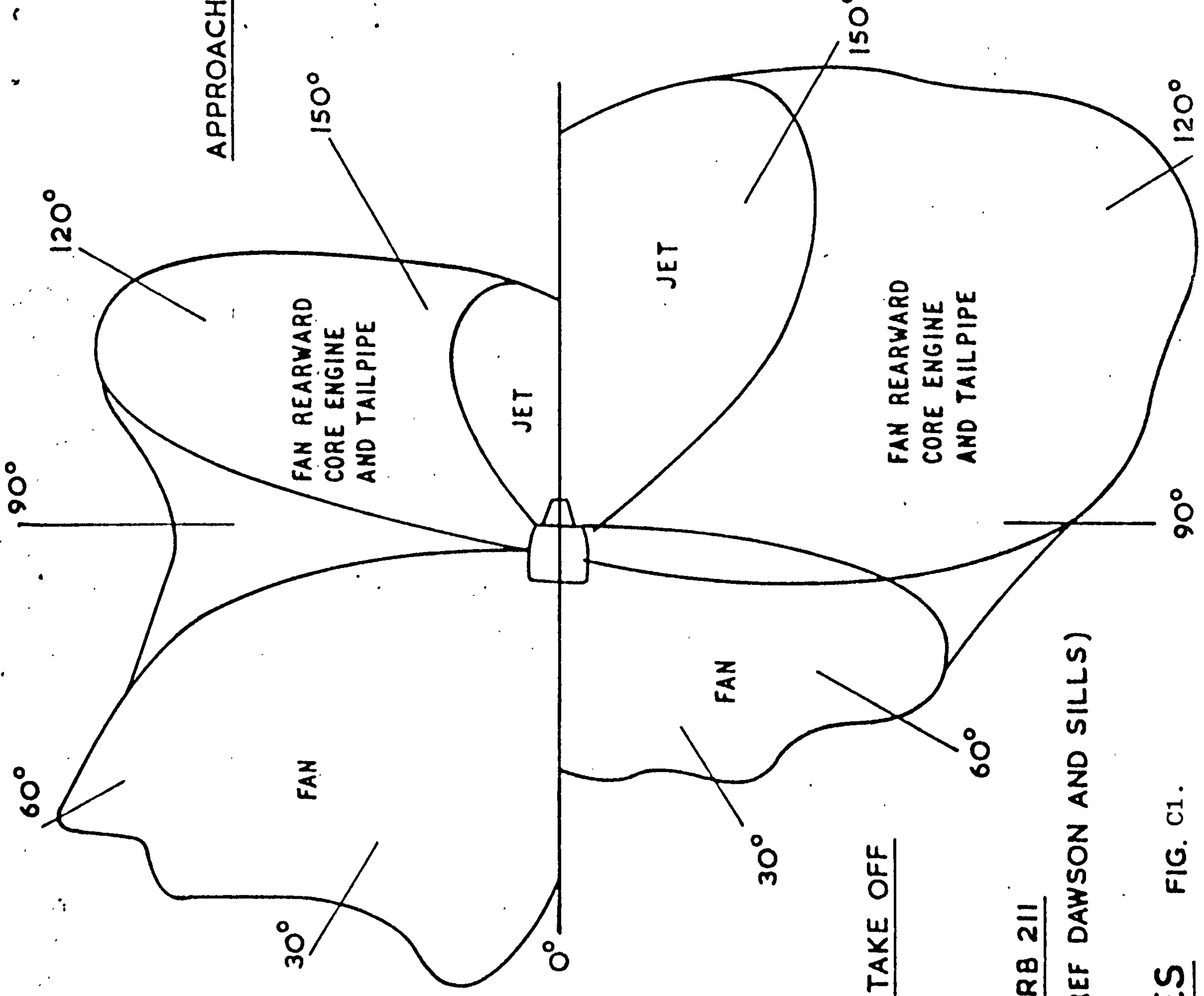
The over the wing blowing concept is relatively more recent than other powered lift systems and hence specific data is less readily available. The shielding effect of the wing and the possibility of acoustic treatment should enable systems of the type to be designed with noise levels comparable to those of the augmentor wing.

C6.5 Comparison of lift systems

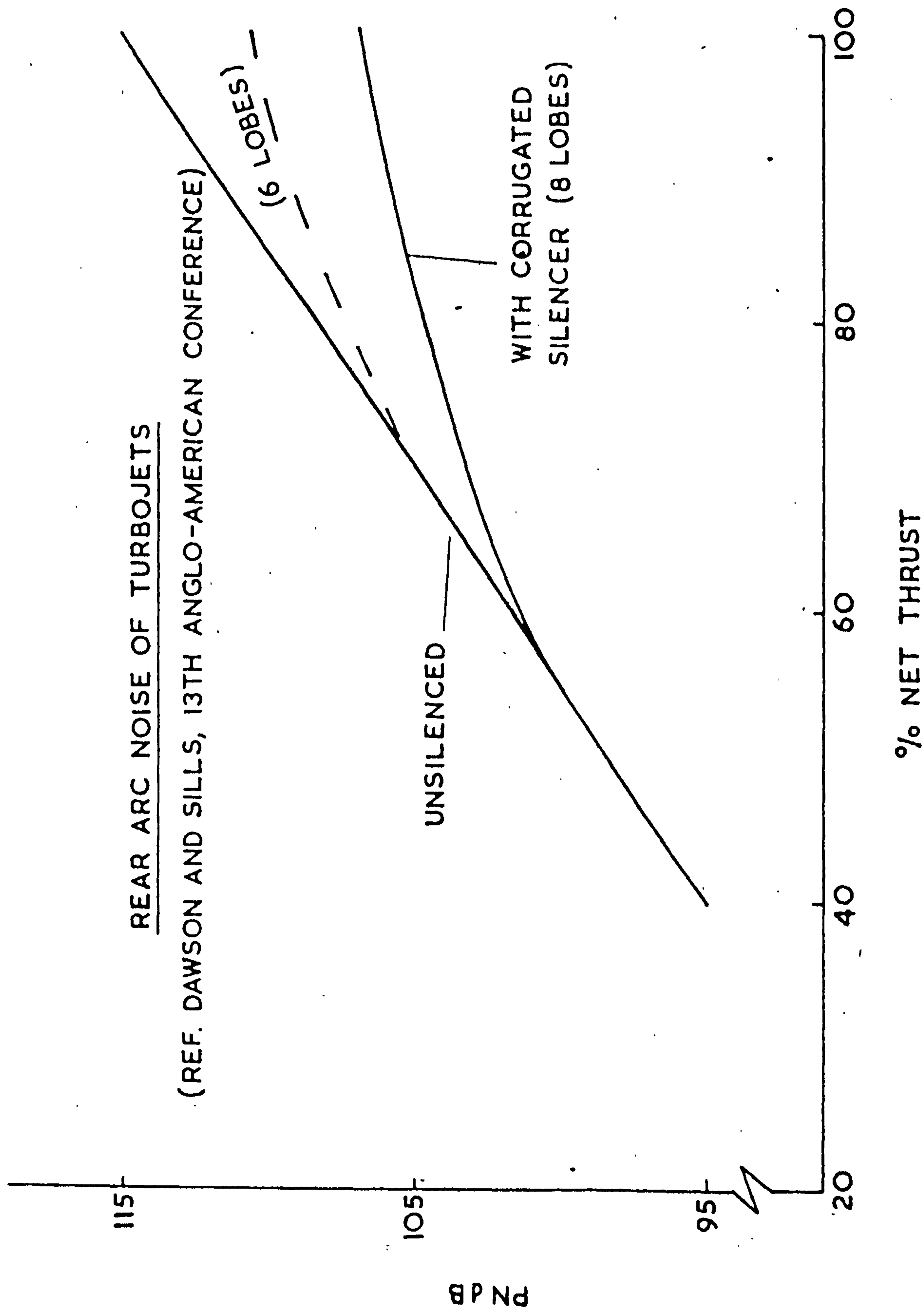
Figure C3, derived from References 53 and 54 compares the noise of the various systems as a function of nozzle pressure ratio.



BY PASS ENGINE



RB 211
(REF DAWSON AND SILLIS)



EFFECT OF JET SILENCERS

NOZZLE AREA 3.45 in^2
FLAP ANGLE 50° (EFFECTIVE)

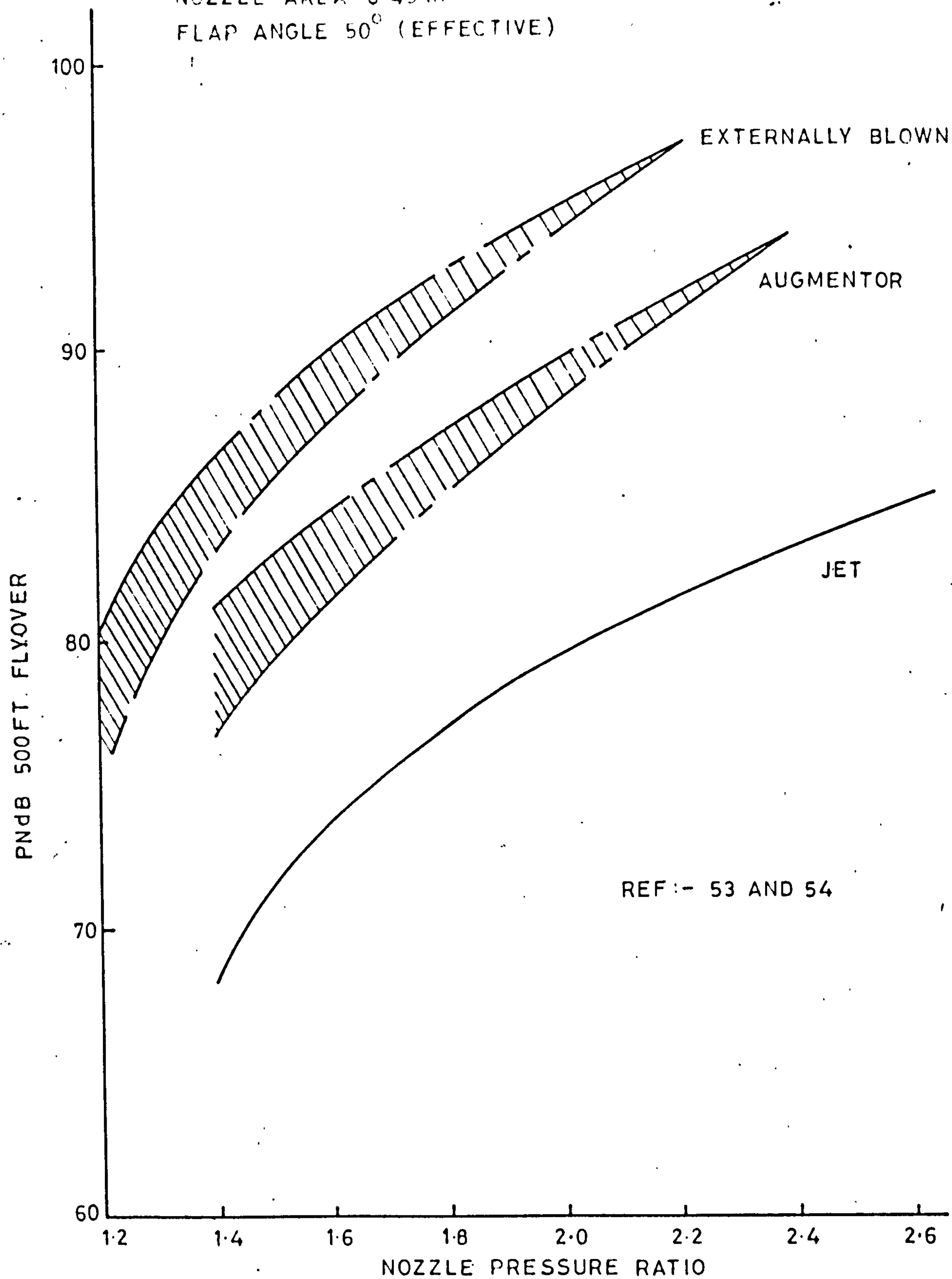


FIG C3 NOISE OF POWERED LIFT SYSTEMS

APPENDIX D

Data for Fan Lift VTOL Aircraft

(This information has been extracted from the project document, DES 1000, prepared by the author and dated August 1970.)

D1. DESIGN REQUIREMENTS

The aircraft was designed to meet B.C.A.R. requirements at the normal take off weight of 125,000 lb. The design value of V_C is 400 knots E.A.S. but was limited to $M = 0.83$ above 16,500 ft. Similarly V_D is 500 knots E.A.S. limited to $M = 0.94$ above 12,000 ft approximately. This is shown in Figure 6.

The airframe was designed to have a life of 40,000 hours with an average flight duration of 30 minutes. The fuselage had to withstand a maximum differential of 8 lb/sq in, the normal cabin altitude not exceeding 8000 ft.

The design, proof, vertical velocity of descent for undercarriage design was 12 ft/sec.

D2. Geometry

D2.1 Wing (See Fig.A70-4)

Gross area	1000 sq ft.
Span	77 ft.
Aspect Ratio	5.9
Leading edge sweepback	28°
Root chord (centreline)	19.32 ft.
Tip chord (nominal)	6.82 ft.
Standard mean chord, \bar{c}	13.07 ft.
Aerofoil sections,	
Root 13% thickness at 0.375c, symmetrical	
Tip 10% thickness at 0.375c, symmetrical	
Linear variation, see Table 3.	
Wing-body angle (centreline chord to body datum)	3°
Dihedral, in wing plane	0°
Location of 0.25 \bar{c} aft of fuselage nose	51.5 ft.
Location of 0.25 \bar{c} aft of centreline leading edge	12.2 ft.
Distance of 0.25 \bar{c} above fuselage datum, centreline of aircraft and wing chord plane	5.53 ft.

D2.2 Trailing edge flaps (See Fig.A70-5)

Type:	Double slotted
Total flap chord/wing chord	0.30
Subsidiary nose flap chord/total flap chord	
upper surface	0.20
lower surface	0.10
High lift flap setting	15° + 15°
High lift and drag flap setting	45° + 15°
Inboard end of flap from aircraft centreline	18.7 ft.
Outboard end of flap from aircraft centreline	30.0 ft.

D2.3 Ailerons (See Fig.A70-5)

Type:	20% control chord nose balance
Aileron chord/wing chord	0.25
Movement	±20°
Inboard end relative to aircraft centreline	30.2 ft.
Outboard end relative to aircraft centreline	38.5 ft.

D2.4	<u>Speed Brake/Spoiler</u> (See Fig.A70-5)	
	Spoiler chord/wing chord	0.1
	Movement relative to local surface	30°
	Leading edge of spoiler aft of wing leading edge	0.66c
	Inboard end relative to aircraft centreline (leading edge)	18.7 ft.
	Outboard end relative to aircraft centreline (leading edge)	30.0 ft.
D2.5	<u>Tailplane</u> (See Fig.A70-4)	
	Gross area	277 sq ft.
	Span	33.3 ft.
	Aspect Ratio	4.0
	Sweepback of leading edge	28°
	Root chord (centreline)	10.42 ft.
	Tip chord (nominal)	6.25 ft.
	Aerofoil section:	
	10% thickness at 0.375c, symmetrical	
	See Table 3.	
	Dihedral	0°
	Movement	+0° up -5° down
	Vertical location of tailplane above fuselage datum	0 ft.
	Distance of centreline chord leading edge aft of fuselage nose	82.7 ft.
D2.6	<u>Elevator</u>	
	Type: Round nose	
	Elevator chord/tailplane chord	0.25
	Movement	+15° down -25° up
	Inboard end of elevator hingeline from aircraft centreline	2.1 ft.
D2.7	<u>Fin</u> (See Fig.A70-4)	
	Nominal area above fuselage datum	280 sq ft.
	Area above fuselage cross section- aerodynamic reference area	205 sq ft.
	Height above fuselage datum	22.9 ft.
	Aspect ratio based on area above fuselage datum	1.87
	based on area above fuselage	1.63

Fin continued

Chord on fuselage datum 17.1 ft.

Tip chord (nominal) 7.4 ft.

Sweepback of leading edge 35°

Aerofoil section:

10% thickness at 0.375c, symmetrical

See Table 3.

Distance of leading edge intersection with fuselage datum from fuselage nose 71.9 ft.

D2.8 Rudder

Type: Round nose

Rudder chord/fin chord 0.4

Height of rudder root above fuselage datum 11.66 ft.

Movement ±20°

D2.9 Fuselage (See Fig.A70-2)

Overall length 94.2 ft.

Maximum diameter 12.5 ft.

Maximum cabin internal width 11.65 ft.

Cabin height on centreline 6.5 ft.

Cabin length, overall 67.3 ft.

D2.10 Lift Engine Nacelles (See Fig.A70-3)

Overall length 65.8 ft.

Maximum width 8.0 ft.

Maximum depth 5.0 ft.

Distance of nacelle nose aft of fuselage nose 19.3 ft.

Distance of nacelle centreline from aircraft centreline 14.5 ft.

Height of nacelle datum above bottom of nacelle 1.67 ft.

Height of nacelle datum above fuselage datum 4.67 ft.

D2.11 Undercarriage

Type: Nosewheel

Wheelbase 43.8 ft.

Track (to centre of mainwheel unit) 13.3 ft.

Main Undercarriage unit

4 wheel side by side arrangement

Tyres: 34 in dia. x 9.25 in wide - 16 in rim.

Tyre pressure 150 p.s.i.

Track over inner pair of wheels 3.35 ft.

outer pair of wheels 7.35 ft.

Static tyre closure 0.25 ft. approx.

Maximum tyre closure 0.5 ft. "

Location of main undercarriage leg aft
of fuselage nose 4.58 ft.

Nosewheel unit.

Twin wheels

Tyres: 34 in dia. x 9.25 in wide - 16 in rim.

Tyre pressure 150 p.s.i.

Wheel track 1.5 ft.

D3 Power Plants

D3.1 Propulsion Engines

Type: Three shaft bypass jet engine

Installation: 2 fin mounted pods

Bypass ratio 3

Sea Level Static Thrust 21,500 lb.

Overall length of pod 12.5 ft.

Intake diameter 4.6 ft.

Maximum diameter of pod 5.6 ft.

Location of engine centreline above
fuselage datum 9.25 ft.

Location of engine centreline from aircraft
centreline 3.58 ft.

Location of pod front face aft of fuselage
nose 80.0 ft.

D3.2 Lift Engines

Type: RB 202 fan engine derivative

Installation: 2 nacelles each with 6 engines.

Sea level static thrust 14,500 lb.

Overall diameter 7.3 ft.

Overall depth 3.7 ft.

Distance of mounting trunnion from intake face 2.92 ft.

Lift Engines continued

Fore and aft location of engines
- see Fig. A70-3

Tilt angle, fore and aft

$\pm 25^{\circ}$ aft
 $\pm 15^{\circ}$ forward

D3.3 Auxiliary Power Unit

Type:

Location of A.P.U. above fuselage datum 3.35 ft.

Location of A.P.U. front face aft of fuselage nose 83.0 ft.

D4. Weights, Centres of Gravity and Moments of Inertia

Design normal weight at take off 125,000 lb.

Maximum landing weight 120,000 lb.

Minimum flying weight 80,000 lb.

As prepared for service weight 78,950 lb.

Maximum payload 24,000 lb.

Maximum fuel load 35,000 lb.

Weight breakdown - see Table 1

Centre of Gravity at A.P.S. Weight, relative to 0.25c and fuselage datum:-

Undercarriage Retracted \bar{x} = 2.30 ft aft

\bar{z} = 3.57 ft above

Undercarriage Extended \bar{x} = 2.20 ft aft

\bar{z} = 3.45 ft above

Allowable centre of gravity range 1.0 ft forward to 1.7 aft of datum

Moments of Inertia - see Table 2.

D5. Aerodynamic Information

Maximum lift coefficients (untrimmed):-

Basic wing 1.2

Flaps at high lift setting, see 3.2 1.65

Flaps at high drag setting, see 3.2 1.8

Normal flight maximum (1.22 V_S) 1.2

Drag polars:-

Cruise at $M = 0.80$ and 20,000 ft. $C_D = 0.0297 + 0.091C_L^2$
 $M = 0.4$ and sea level $C_D = 0.0245 + 0.066C_L^2$

Take off, high lift configuration, sea level
 $C_D = 0.083 + 0.06(C_L - \Delta C_L)^2 + 0.19\Delta C_L^2$

Landing, high drag configuration, sea level
 $C_D = 0.127 + 0.06(C_L - \Delta C_L)^2 + 0.19\Delta C_L^2$
 (where ΔC_L is the increment due to flap)

Pitching moment coefficient

Clean aircraft	-0.09
High lift configuration	-0.26
High drag configuration	-0.35

Forward movement of aerodynamic centre due to
 body and nacelle effects 1.45 ft.

Location of low speed overall wing-body aero.
 centre from fuselage nose 50.4 ft.

Spanwise variation of wing aero. centre. See Fig.A70-11

Location of mean tailplane aero. centre aft of
 fuselage nose 88.7 ft.

Spanwise variation of tailplane aero. centre. See Fig.A70-14

Location of mean fin aero. centre aft of
 fuselage nose 85.0 ft.

Height variation of fin aero. centre. See Fig.A70-16

Wing no lift angle, clean, relative to wing
 centreline chord -1°

Slope of wing-body lift curve, a_1 . See Fig.A70-8.

Slope of aileron hinge moment curve due to
 wing incidence, b_1 -0.27

Slope of aileron hinge moment curve due to
 aileron angle, b_2 -0.68

Slope of tailplane lift curve, a_{1T} . See Fig.A70-8

Ratio of elevator lift curve slopes, a_{2T}/a_{1T} 0.61

Slope of elevator hinge moment curve due to
 tailplane incidence, b_{1T} -0.22

Slope of elevator hinge moment curve due to
 elevator angle, b_{2T} -0.64

Slope of fin lift curves, a_{1F} (net) a_{1FT} (with
 body and tail effects) See Fig.A70-8.

Ratio of rudder lift curve slopes a_{2F}/a_{1F} 0.31

Slope of rudder hinge moment curve due to fin incidence, b_{1F} -0.32

Slope of rudder hinge moment curve due to rudder angle, b_{2F} -0.61

Slope of lift decrement due to operation of spoilers/speed brakes -0.19

Slope of drag increment due to operation of spoiler/airbrakes 0.014

Downwash angle at tailplane. See Fig.A70-10

Rolling moment coefficients:-

Due to aileron angle, l_{ξ} } See Fig.A70-9

Due to rolling, l_p

Due to sideslip l_v

Due to yawing l_r

$$- \left[0.060 + 0.14C_L + (0.033 - 0.061C_L) \right] a_{1BT}$$

$$a_{1BT}(0.03 - 0.055C_L) + 0.21C_L$$

Yawing moment coefficients:-

Due to sideslip n_v $0.13a_{1BT} - 0.09$

Due to yawing n_r $- \left[0.07 + 0.115a_{1BT} + 0.018C_L^2 \right]$

Side force coefficient due to sideslip, y_v $- \left[0.3 + 0.28a_{1BT} \right]$

Tailplane rolling moment coefficient due to sideslip, K_{β} 0.15

(Note all derivatives are based on the reference dimensions and areas quoted a paragraph 3. Hinge moment coefficients only are based on control surface area and chord aft of the hinge line. All angular measure is in radians unless otherwise stated).

D6 Load Distributions

D6.1 Aerodynamic Loads

The wing spanwise load distributions due to incidence, flaps, ailerons and spoiler deflection are shown in Figures A70-12 and 13. The tailplane spanwise load distribution due to both incidence and control deflection is given in Figure A70-15 whilst the corresponding information for the fin and rudder appear in Figure A70-17 and 18. Figure A70-19 gives the lift distribution along the fuselage and lift engine nacelle.

The chordwise load distributions due to control, flap and spoiler deflections are shown in Figures A70-20 and 21. Clean aerofoil chordwise loading may be evaluated by the method given in DRS 545.

D6.2 Inertia Loads

The longitudinal distribution of loads which make up the 'As prepared for service weight' is given in Figure A70-22. This diagram also shows the fuselage longitudinal load distributions. The corresponding distribution for the lift engine nacelle is given in Figure A70-23.

The spanwise inertia distributions for the wing, tailplane and fin are covered by Figures A70-24 to 26 respectively. In converting these to the longitudinal distribution the chordwise inertia positions were assumed to be located at approximately 35% of the chord.

[It should be noted that each of these inertia distributions contributes only one of the two orthogonal components which make up the moment of inertia of the aircraft. Although this component is the dominant one errors in balance of the aircraft can be anticipated unless a correction factor is applied. The appropriate factors for the APS weight are 1.05, 1.2 and 1.23 for pitch, roll, and yaw motions respectively. At the A.U.W. (108 passengers) the corresponding values are 1.06, 1.15 and 1.39 respectively. In the case of the yaw balance the large effect is mainly due to the wing group and should therefore be dealt with as an effect applied locally].

TABLE D1
Component Weights

COMPONENT	WEIGHT lb	A.U.W. %
Wing, including lift nacelle attachment	11150	8.92
Fuselage	11500	9.20
Lift engine nacelle structure and doors	5960	4.77
Tailplane	1880	1.51
Fin, including propulsion engine attachment	1620	1.29
Main undercarriage	2780	2.23
Nose undercarriage	610	0.48
Structure	35500	28.40
Propulsion engines	7000	5.60
Cowlings, nacelle structure, etc.	900	0.72
Engine controls, systems	500	0.40
Propulsion Engines Group	8400	6.72
Lift engines	11620	9.30
Engine controls, systems, etc.	1500	1.20
Lift Engines Group	13120	10.50
Fuel system	1250	1.00
Power supply systems, including APU	4380	3.51
Flying controls	1870	1.49
Deicing system and fire precautions	830	0.66
Air conditioning system	1500	1.20
Systems	9830	7.86

continued.

Table D1continued

COMPONENT	WEIGHT lb	A.U.W. %
Radio and radar installation	1500	1.20
Instruments	600	0.48
Installations	2100	1.68
Sound proofing	1000	0.80
Flight crew furnishings	400	0.32
Cabin furnishings	1880	1.51
Cabin seats	2500	2.00
Miscellaneous cabin items, water system, etc.	1870	1.50
Furnishings	7650	6.13
Basic Operating Weight	76600	61.29
Passenger service items and expendables	1250	1.00
Crew	1100	0.88
As prepared for service weight	78950	63.17
Passengers, 108	21600	17.27
Fuel	24450	19.56
All Up Weight	125000	100.00

TABLE D2

Moments of Inertia

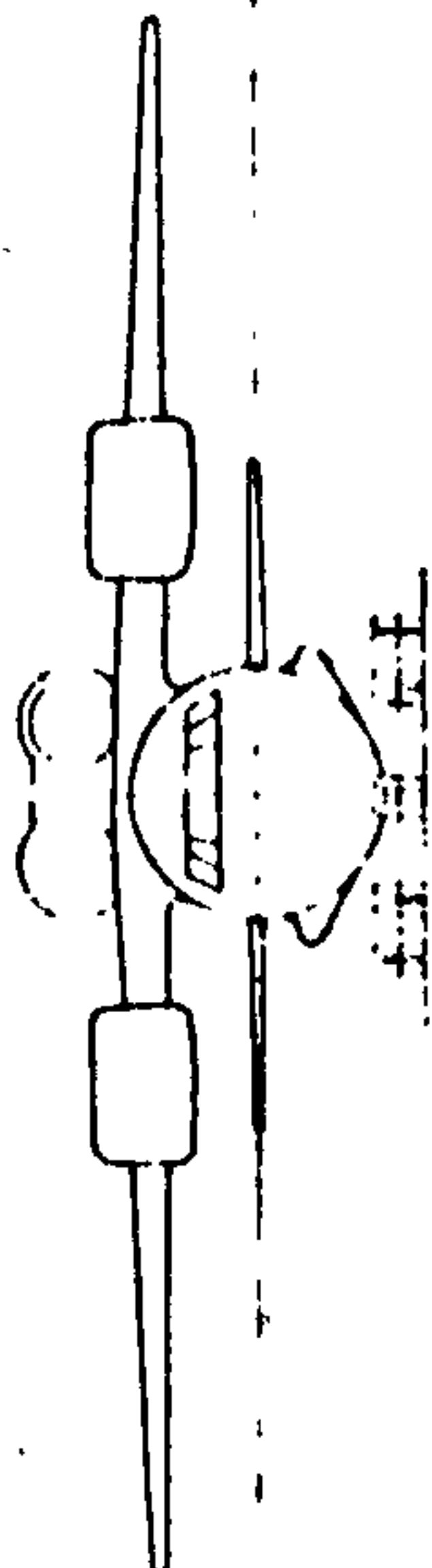
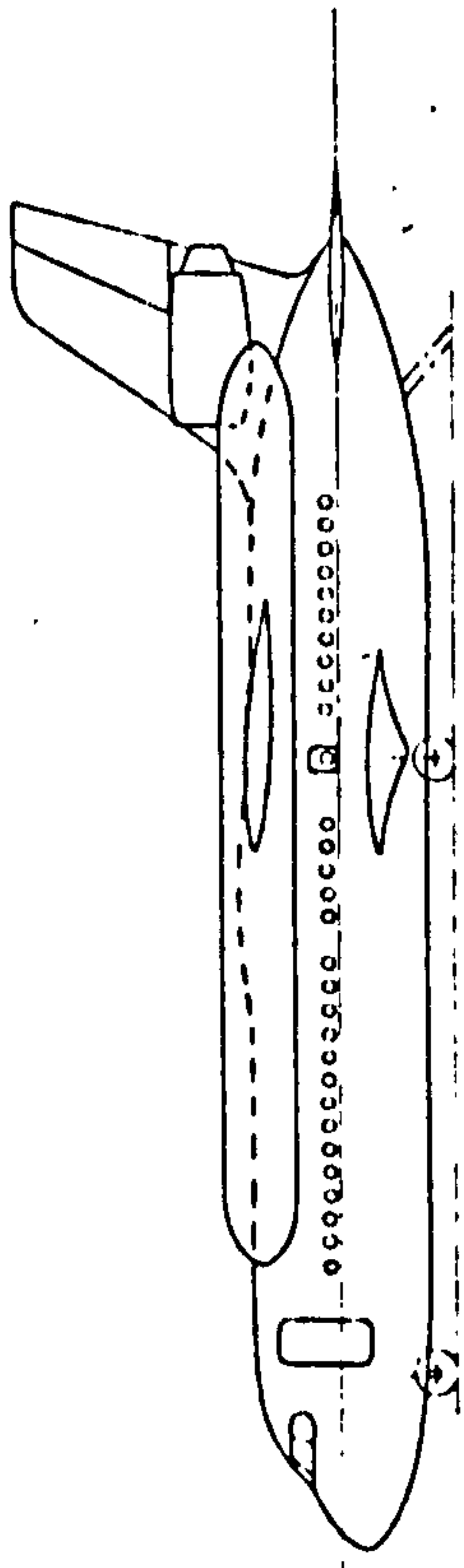
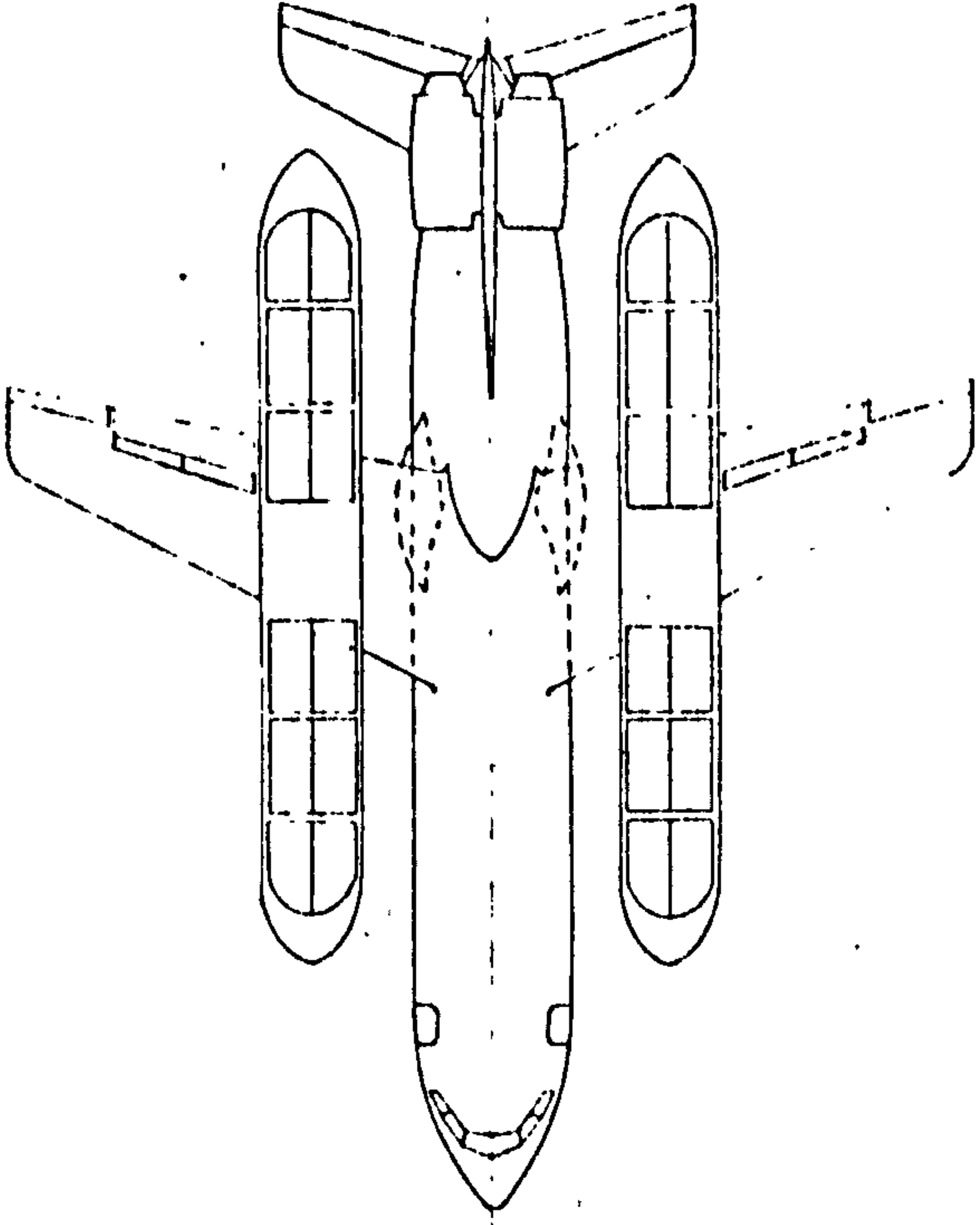
(Relative to 0.3c and 3.5 ft above fuselage datum)

Configuration	Moment of Inertia 10^6 lb ft ²		
	Pitch	Roll	Yaw
As prepared for service 78950 lb	45	12	53
Increment due to 108 passengers 21,600 lb	8	1	7
Increment due to 24450 lb fuel	1	10	11

TABLE D3

Aerofoil Section Ordinates
(10% Thickness Chord Ratio)

% Chord	2% Camber		Symmetrical
	Upper Ordinate	Lower Ordinate	Depth
0.75	1.34	1.02	2.25
1.25	1.89	1.34	2.8
2.5	2.36	1.66	3.85
5.0	3.23	2.21	5.25
7.5	3.86	2.60	6.3
10	4.33	2.76	7.05
15	4.97	3.15	7.75
20	5.43	3.47	9.05
25	5.75	3.63	9.
30	5.98	3.78	9.85
35	6.16	3.87	9.98
40	6.15	3.72	9.95
45	6.06	3.55	9.65
50	5.83	3.31	9.16
55	5.43	2.99	8.4
60	5.04	2.68	7.6
65	4.48	2.21	6.8
70	3.94	1.97	5.75
75	3.39	1.65	4.85
80	2.83	1.34	3.85
85	2.13	0.87	2.95
90	1.42	0.63	2.03
95	0.72	0.30	1.0
100	0	0	0



TITLE - 1970 DESIGN PROJECT			
GENERAL ARRANGEMENT			
DESIGN	DATE	BY	NO.
1970	10/10/70	1970	1
SHEET		OF	
1		1	



TITLE - 1970 DESIGN PROJECT LIFT ENGINE MACELLE LAYOUT	
DRAWN _____	DESIGNED BY _____
A70-3	
0.7130	5-7 25 5411'S

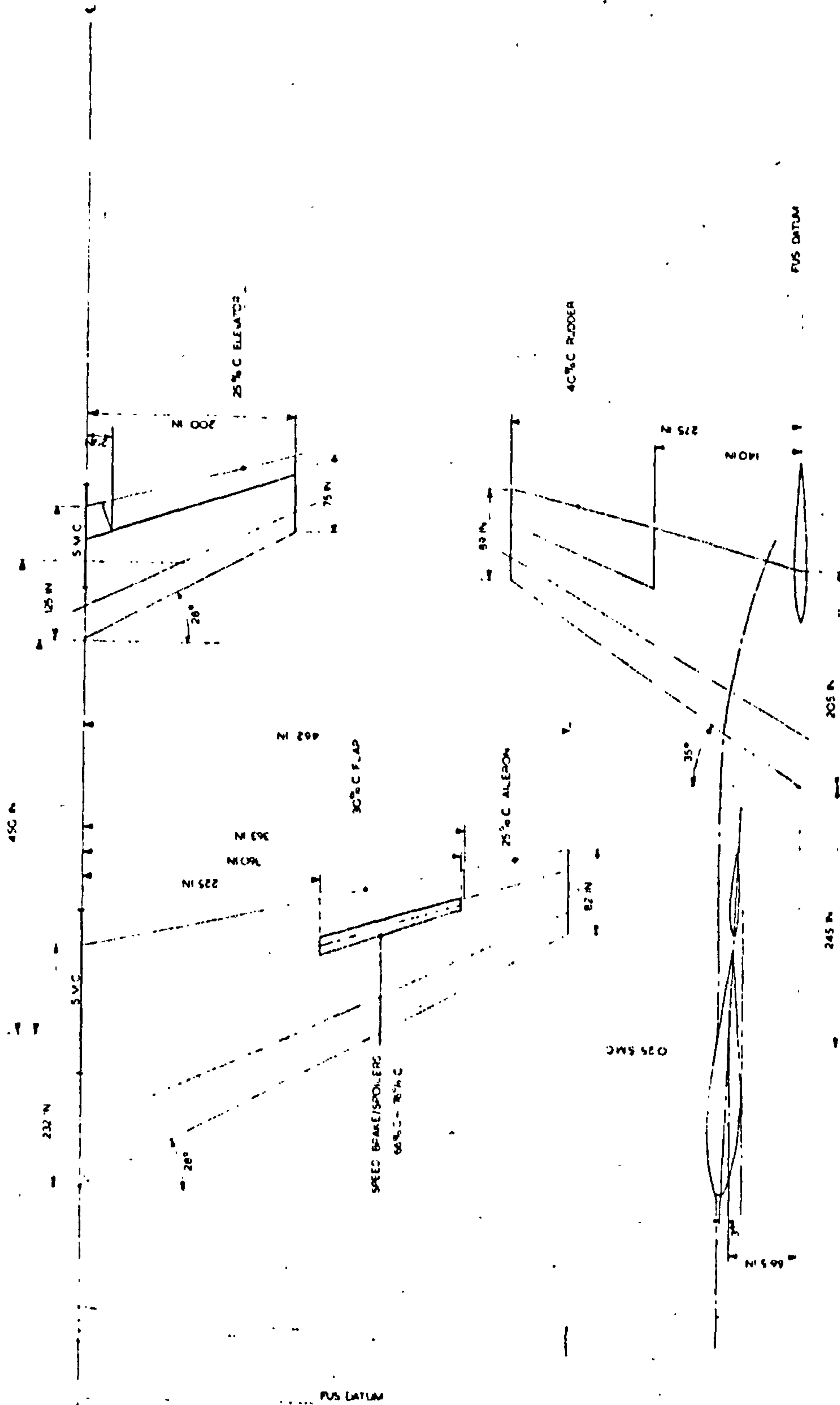
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AEROFOIL GEOMETRY

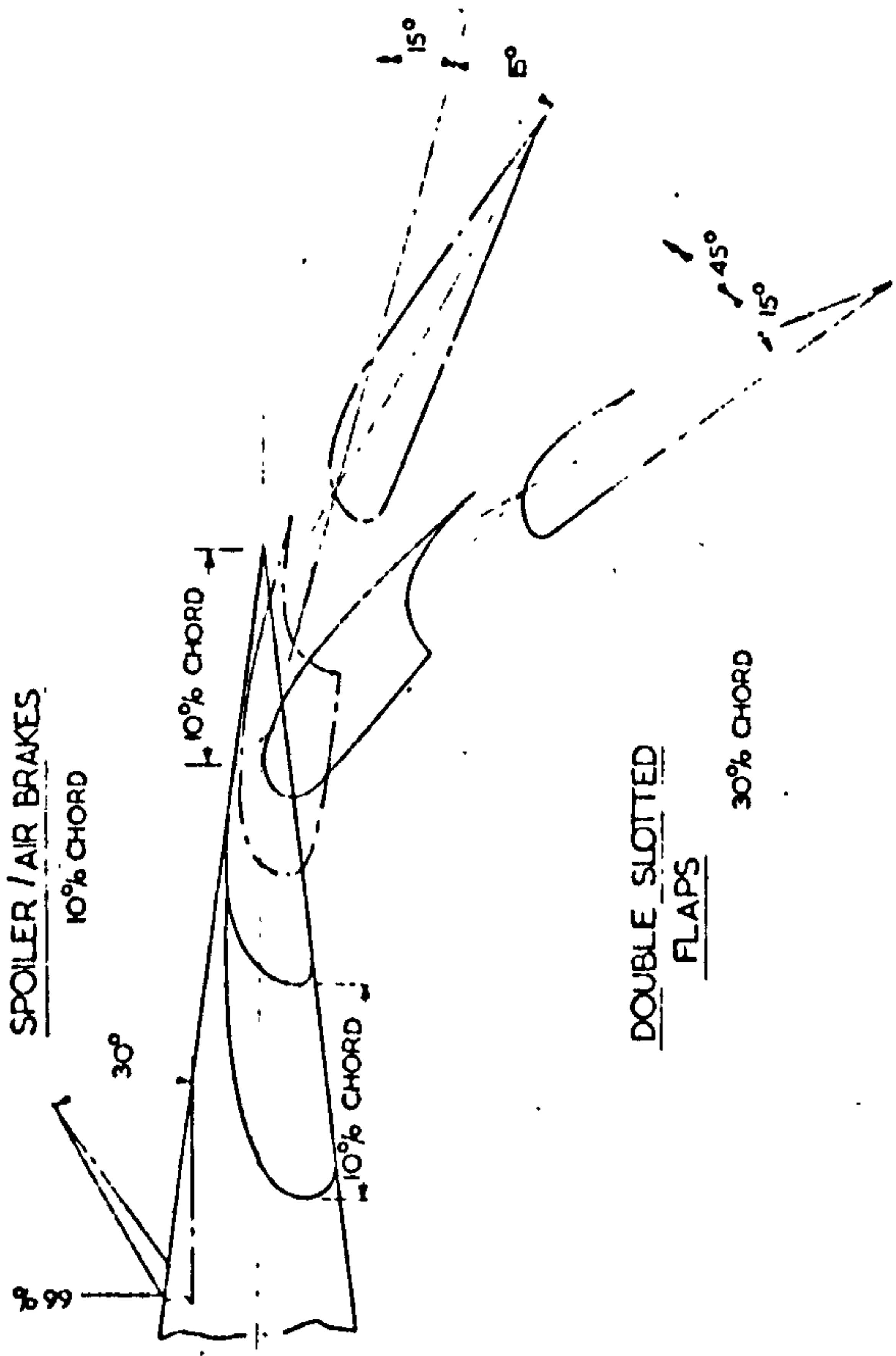
DRAWN BY

DATE

A 70-4

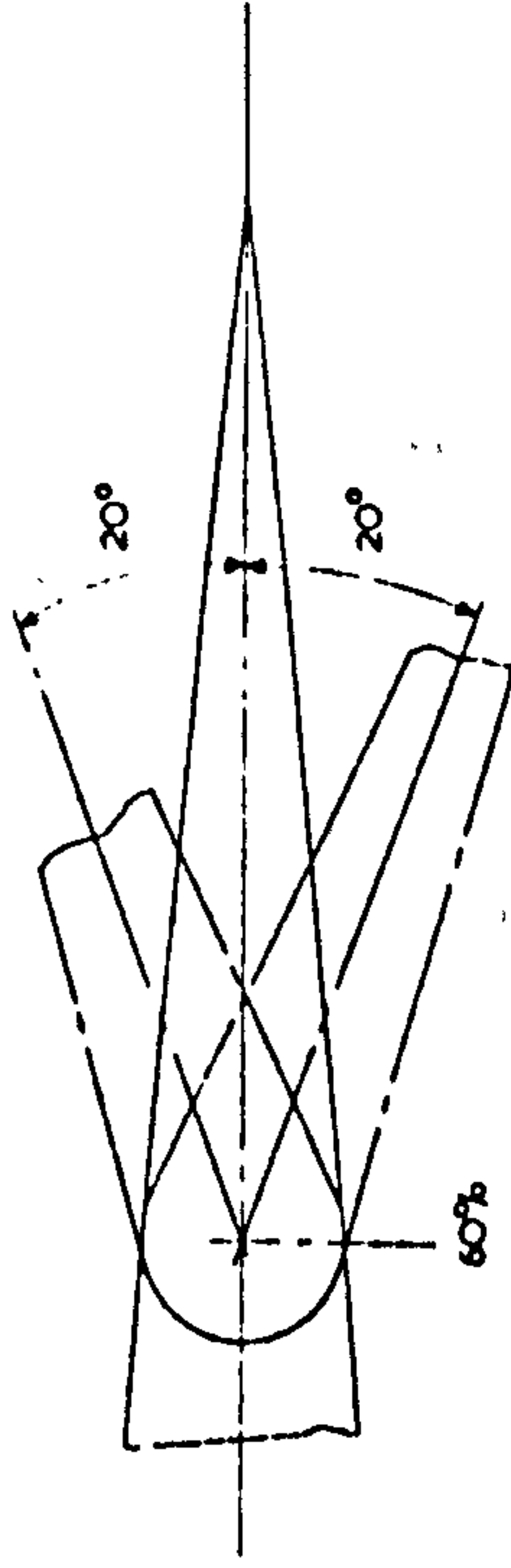
SHEET OF 4



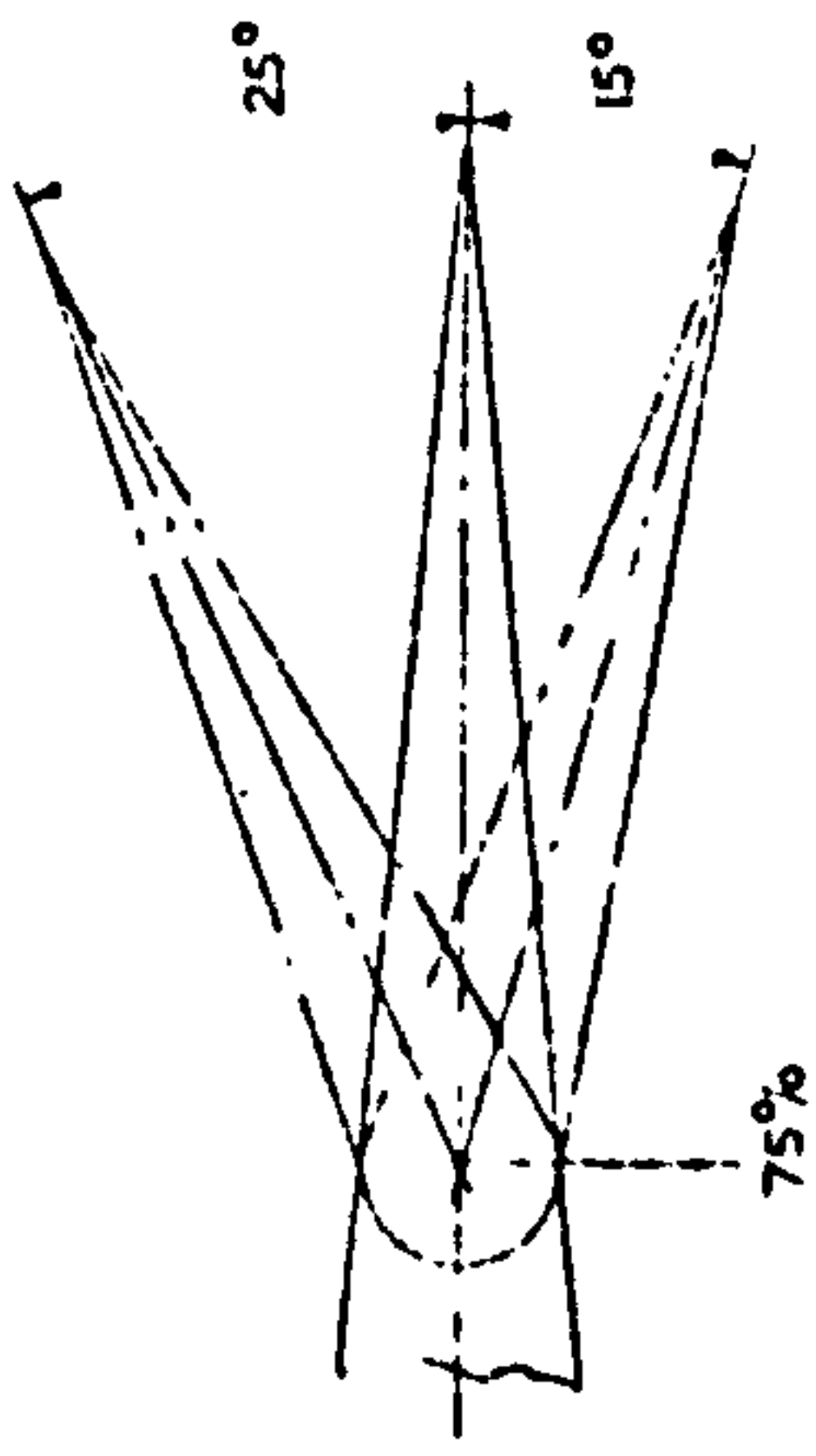


DOUBLE SLOTTED
FLAPS

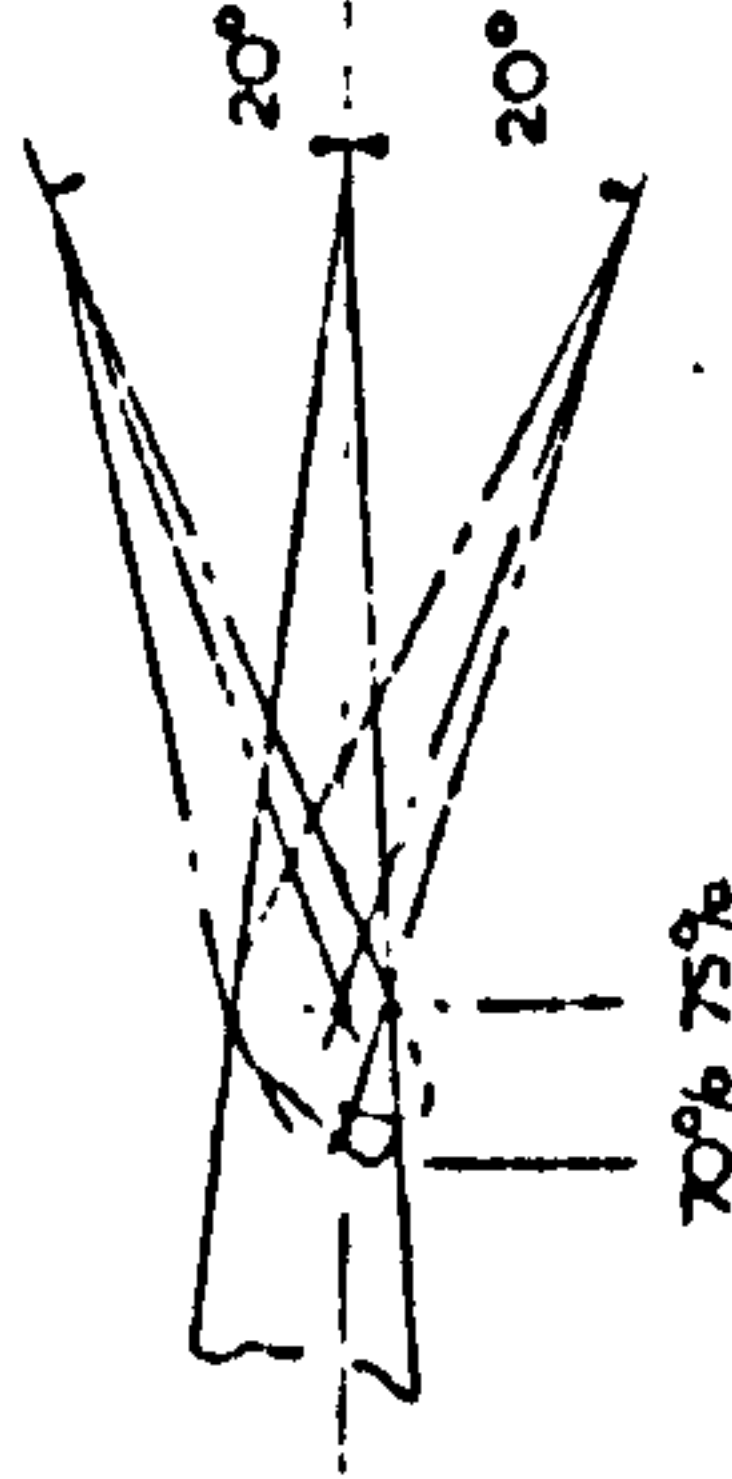
30% CHORD



RUDDER



ELEVATOR



AILERON

TITLE - A70 PROJECT GEOMETRY OF AUXILIARY SURFACES			
DRAWN	DATE	BY	NO.
		A70-5	
1/1/70	15	OF	SHEETS

LEVEL SPEED PERFORMANCE

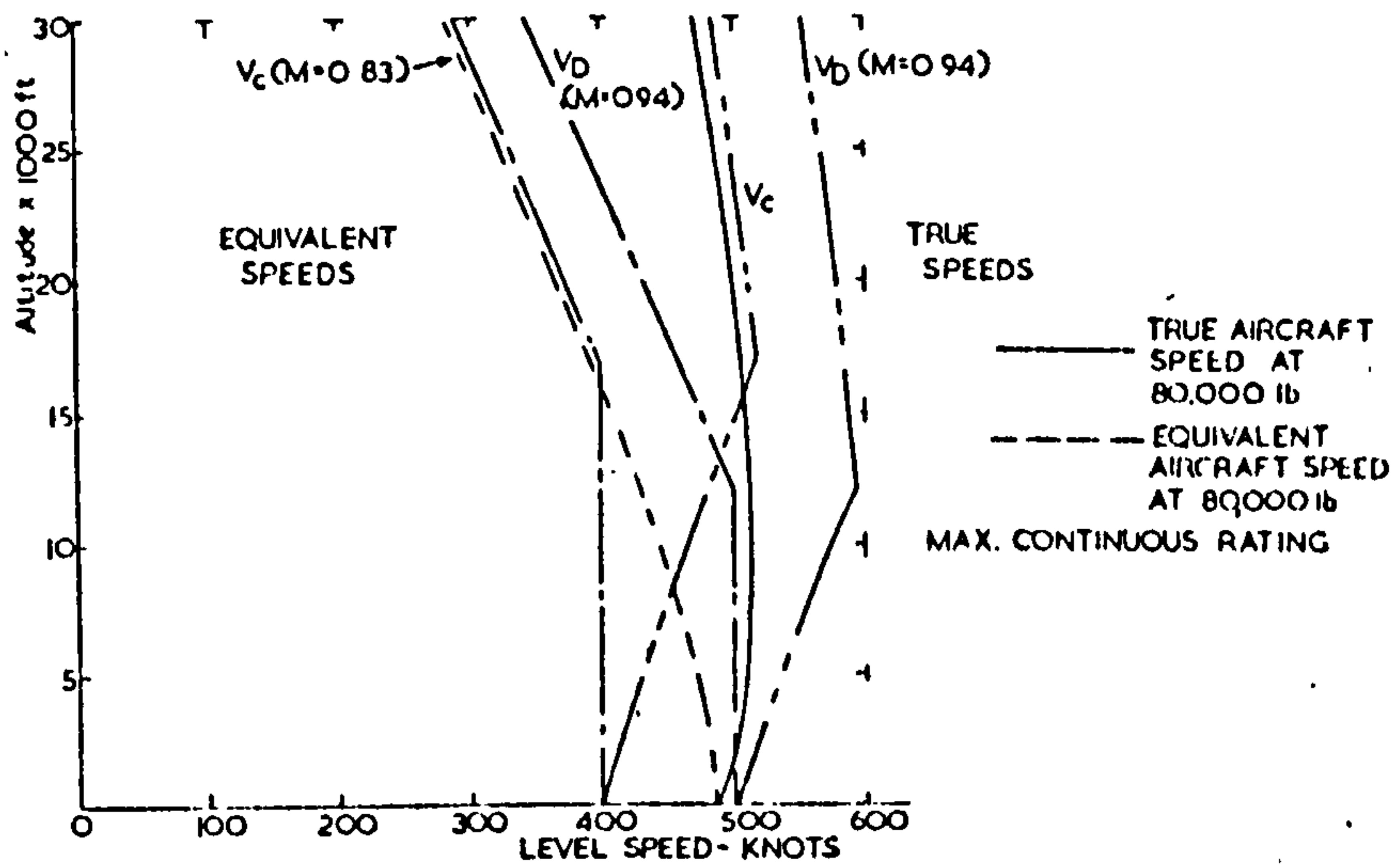


FIG A70-6

PAYLOAD - RANGE PERFORMANCE (STILL AIR VERTICAL TAKE OFF AND LANDING)

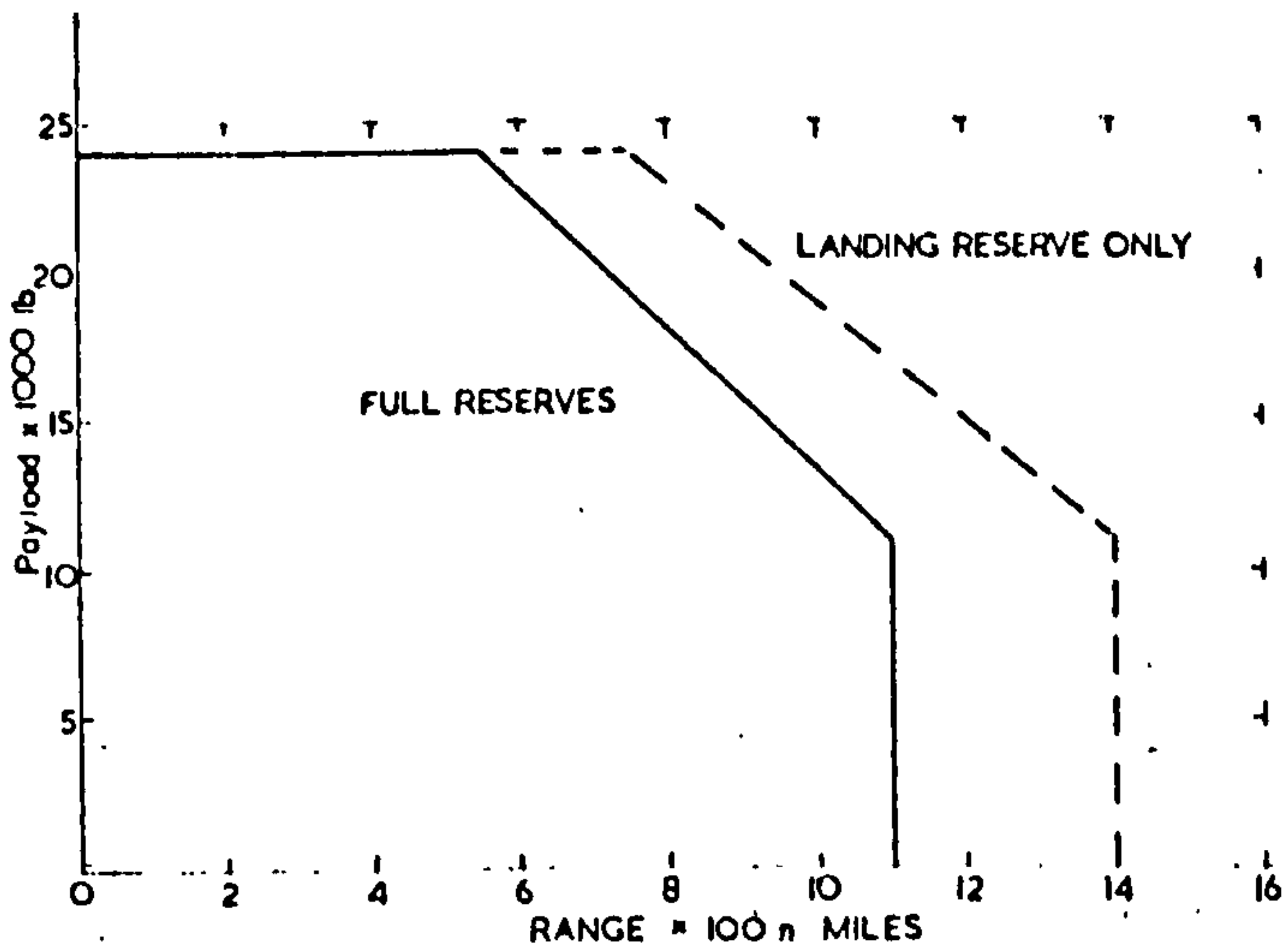


FIG A70-7

LIFT CURVE SLOPES

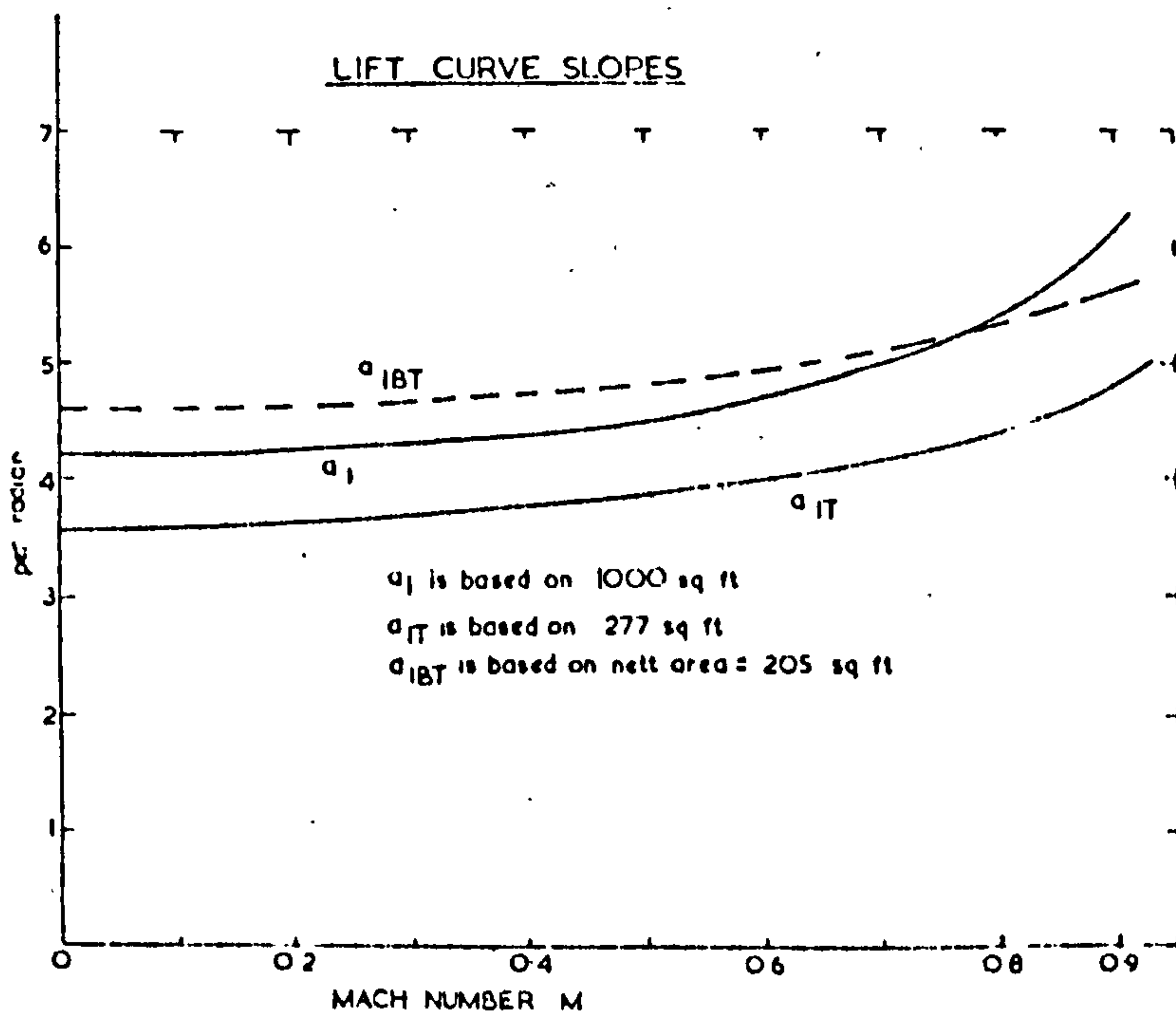
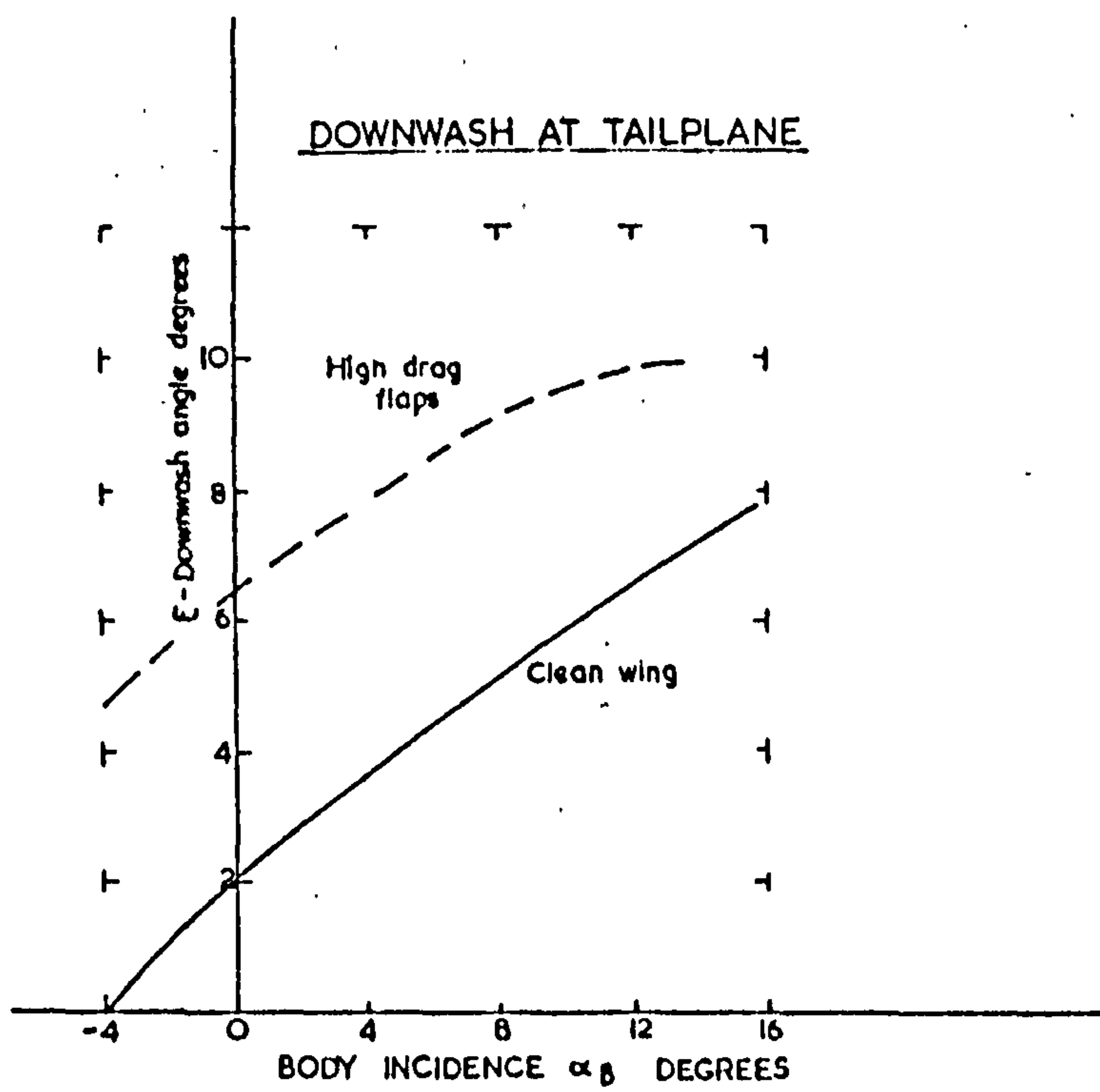
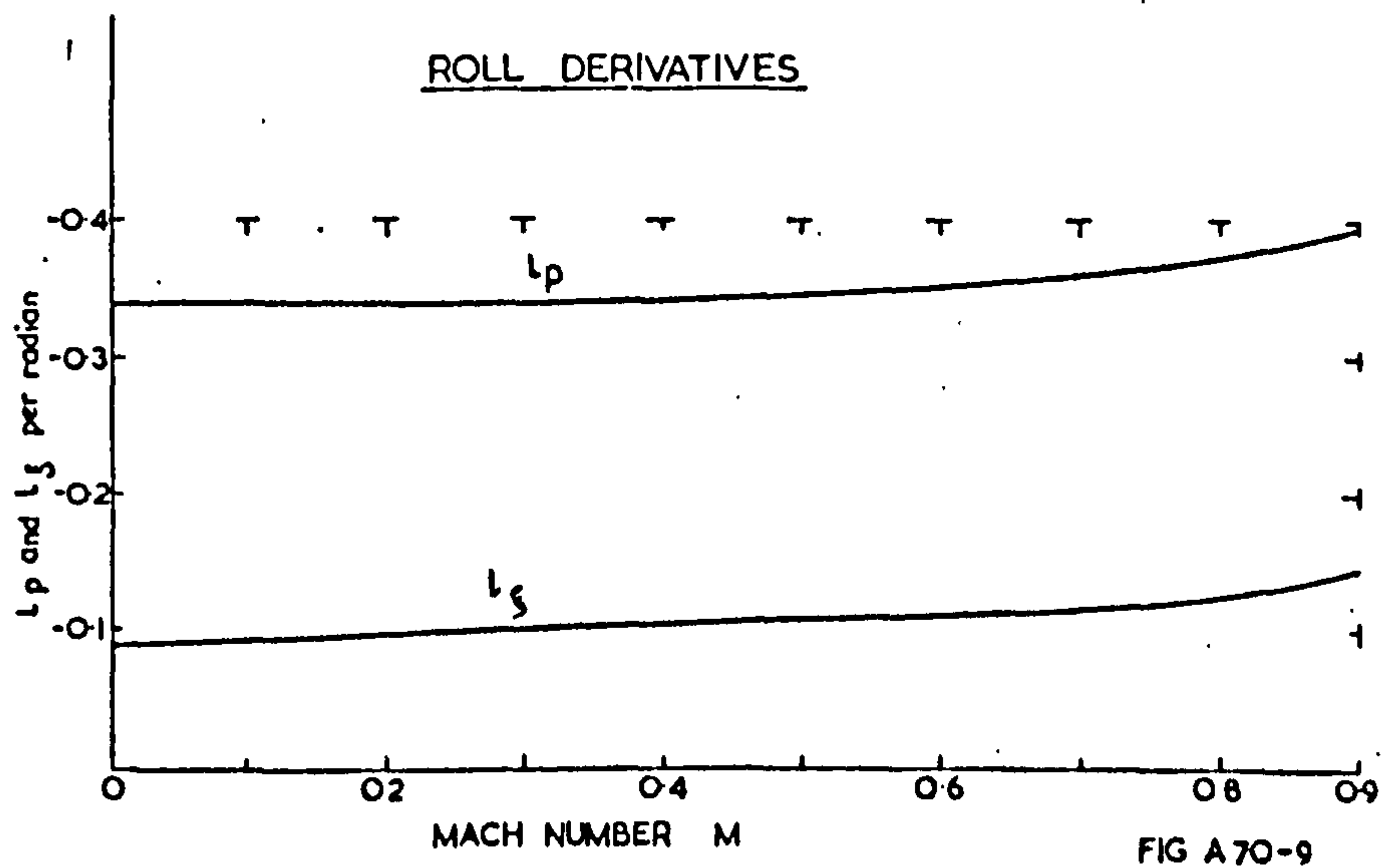


FIG A70-8



SPANWISE VARIATION OF WING AERO CENTRE

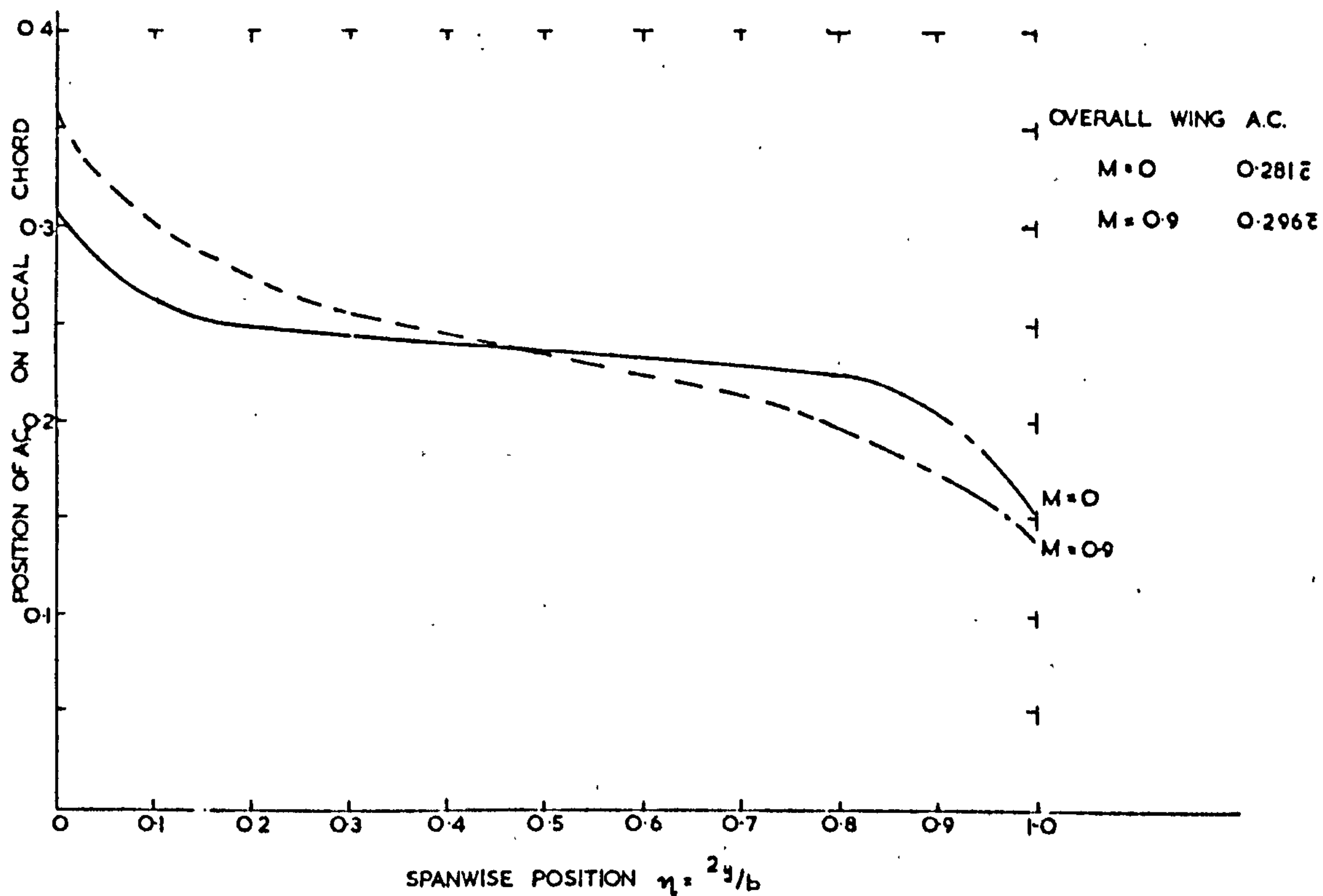


FIG A7D-11

WING SPANWISE AIRLOADING DUE TO INCIDENCE

WING CONTRIBUTION TO TOTAL LIFT (LESS TAIL) 93%

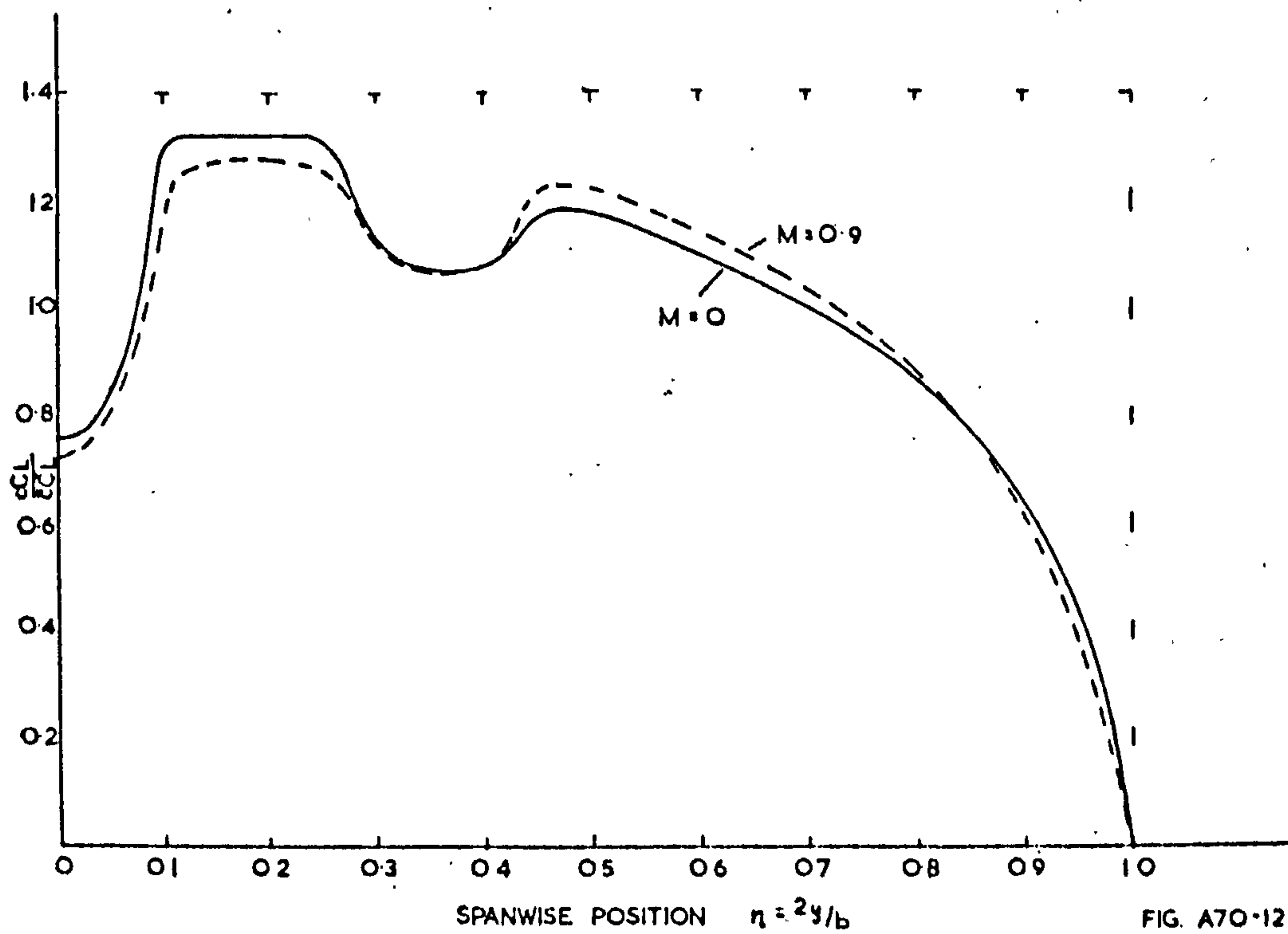
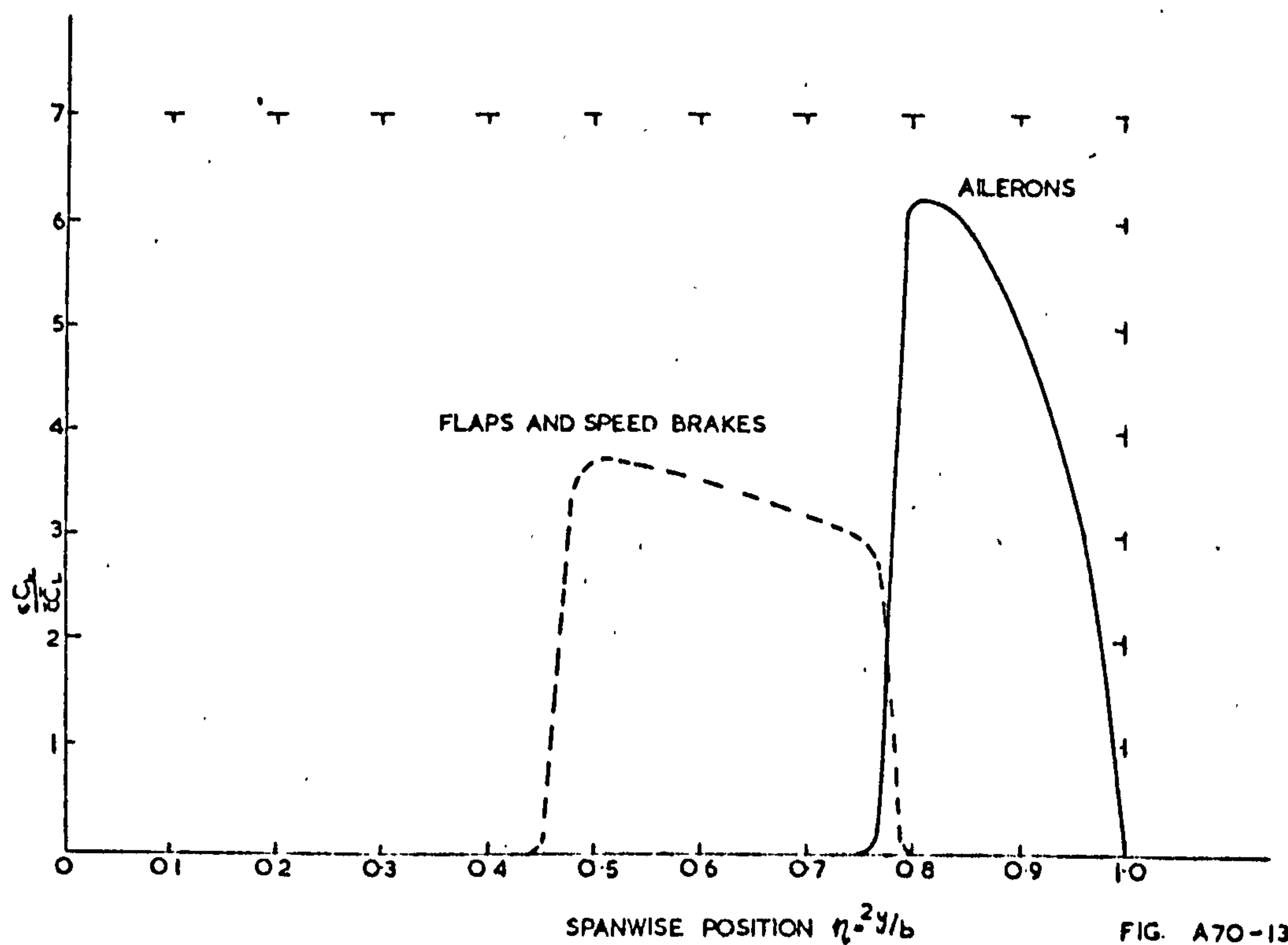
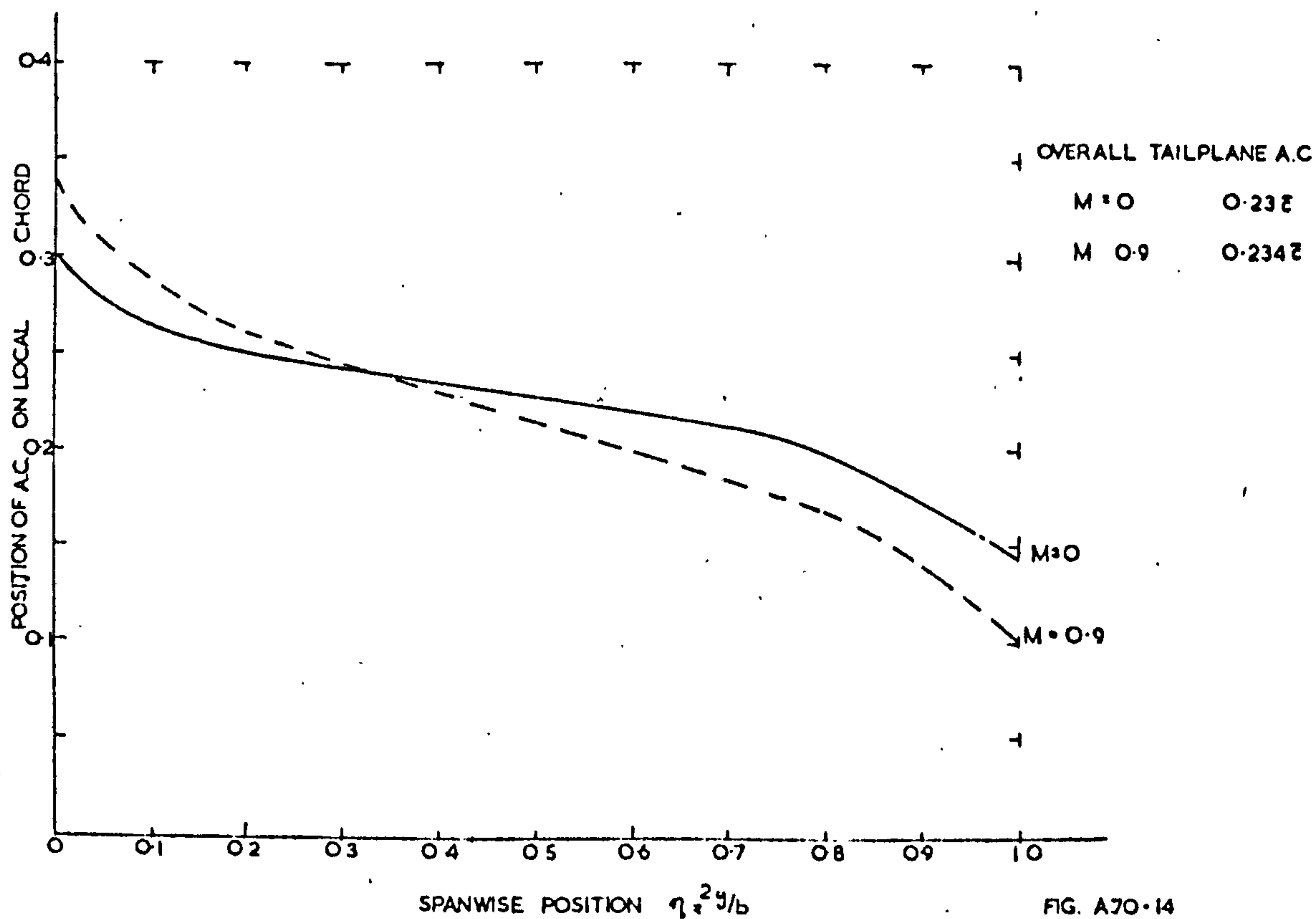


FIG. A7D-12

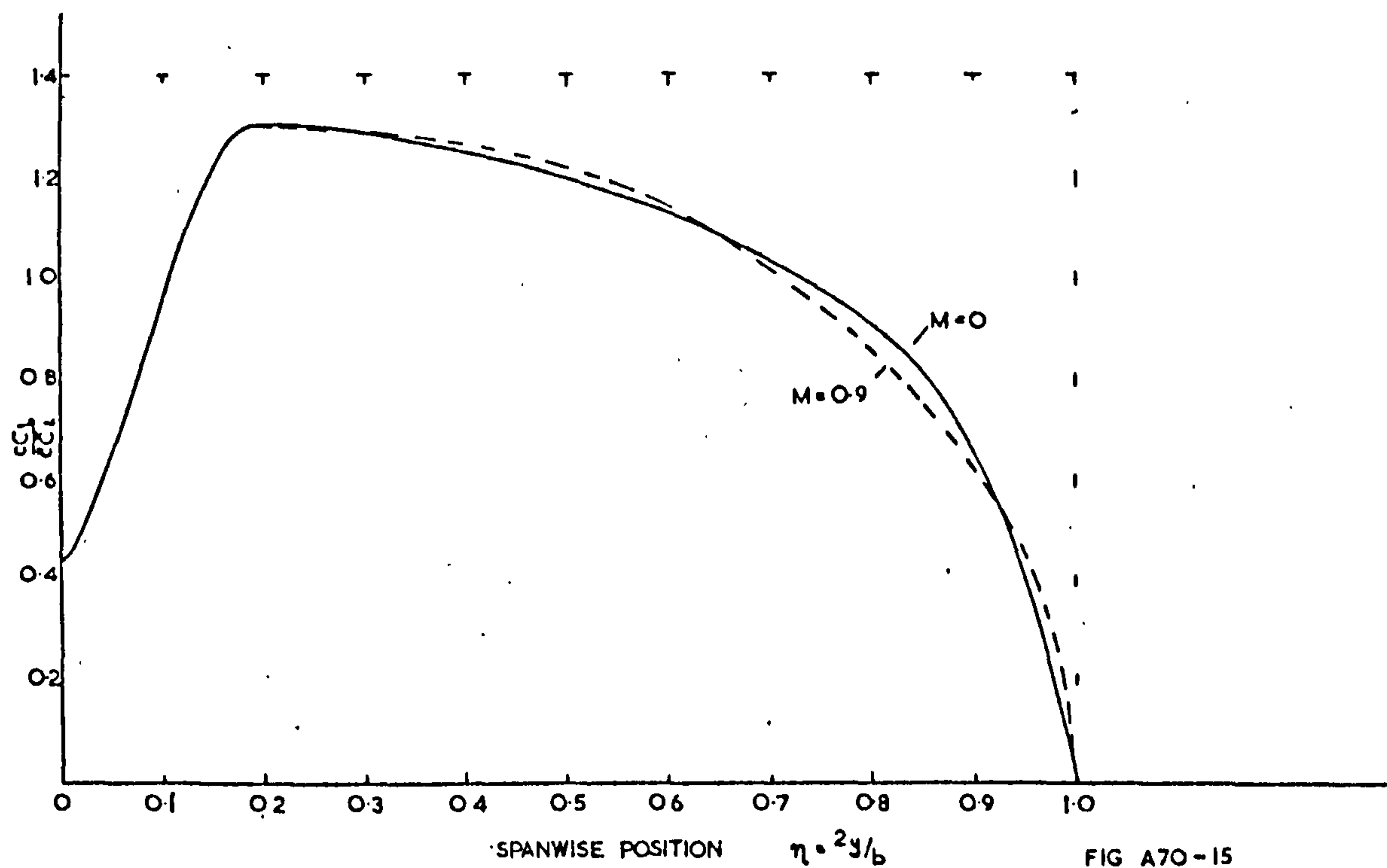
WING SPANWISE AIRLOADING DUE TO AUXILIARY SURFACES



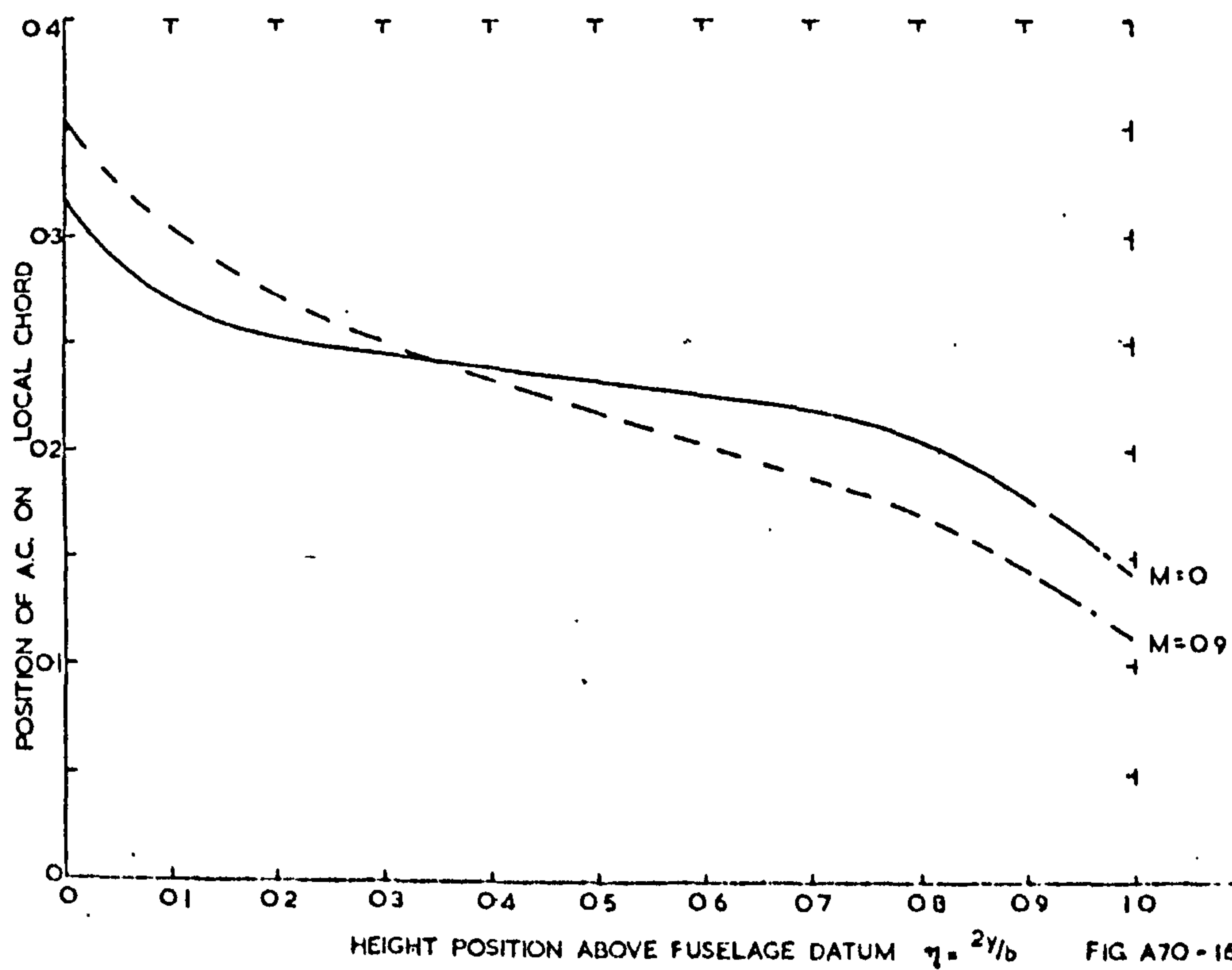
SPANWISE VARIATION OF TAILPLANE AERO CENTRE



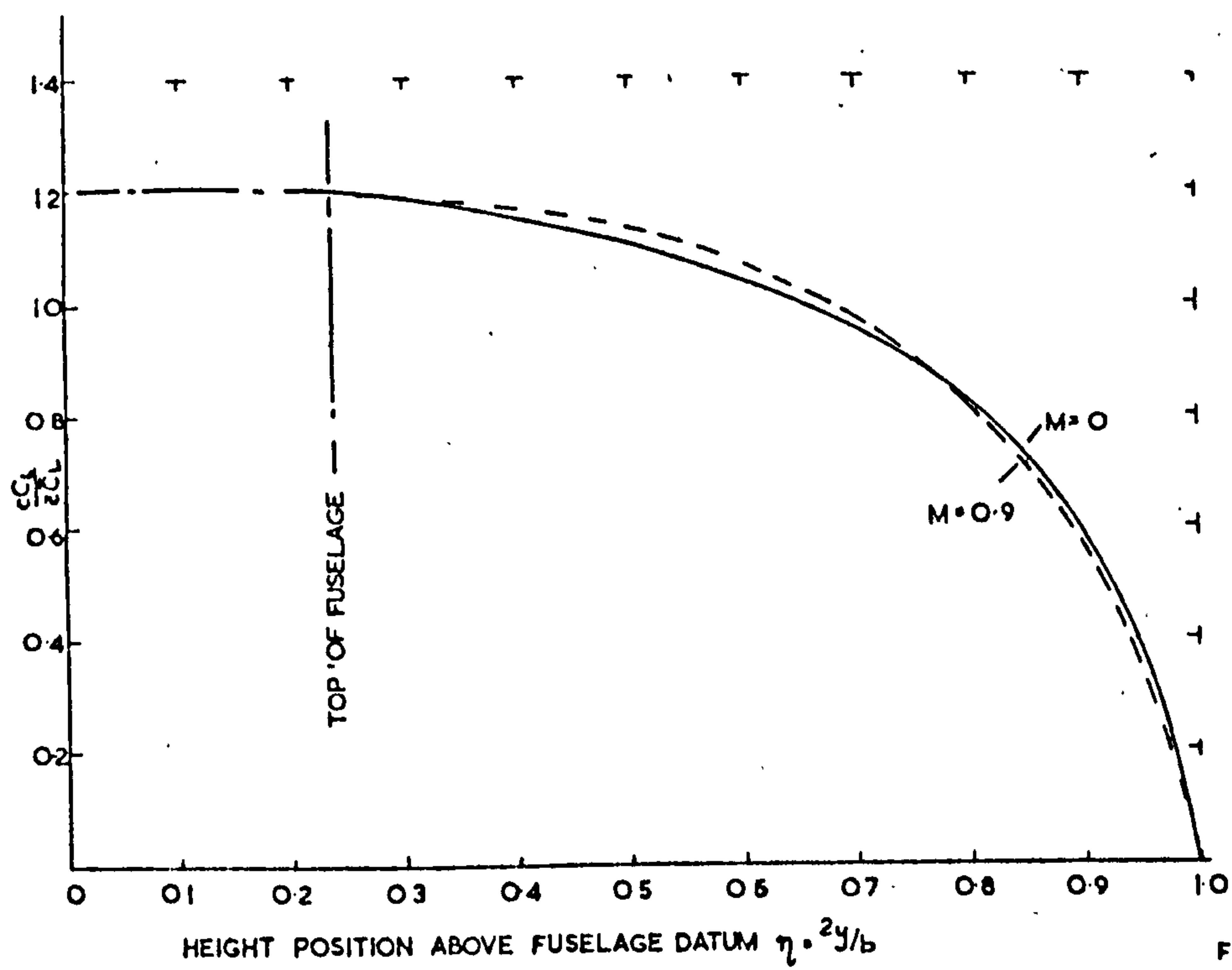
TAILPLANE SPANWISE AIRLOAD DISTRIBUTION DUE TO INCIDENCE AND ELEVATOR



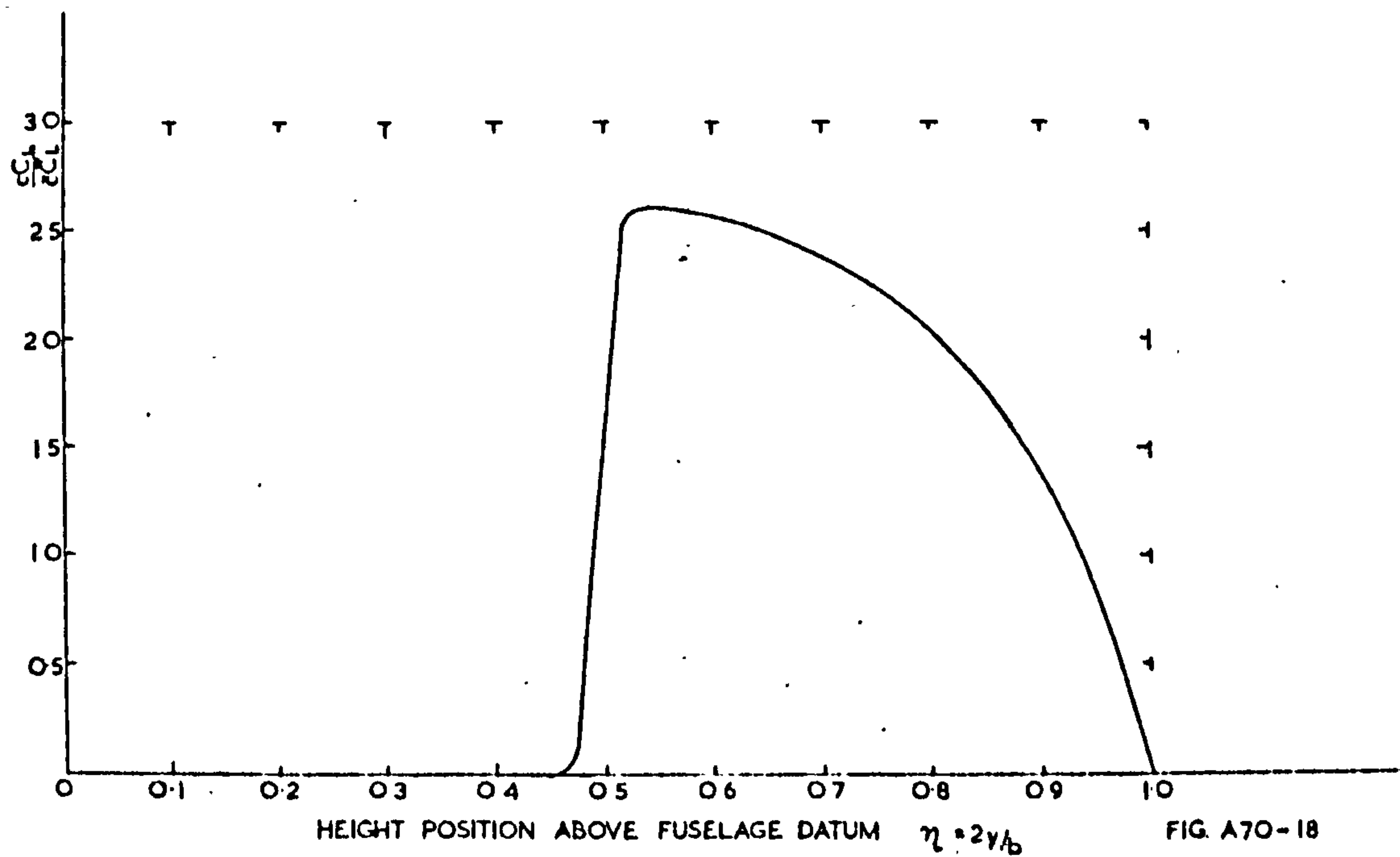
HEIGHT VARIATION OF FIN AERO CENTRE

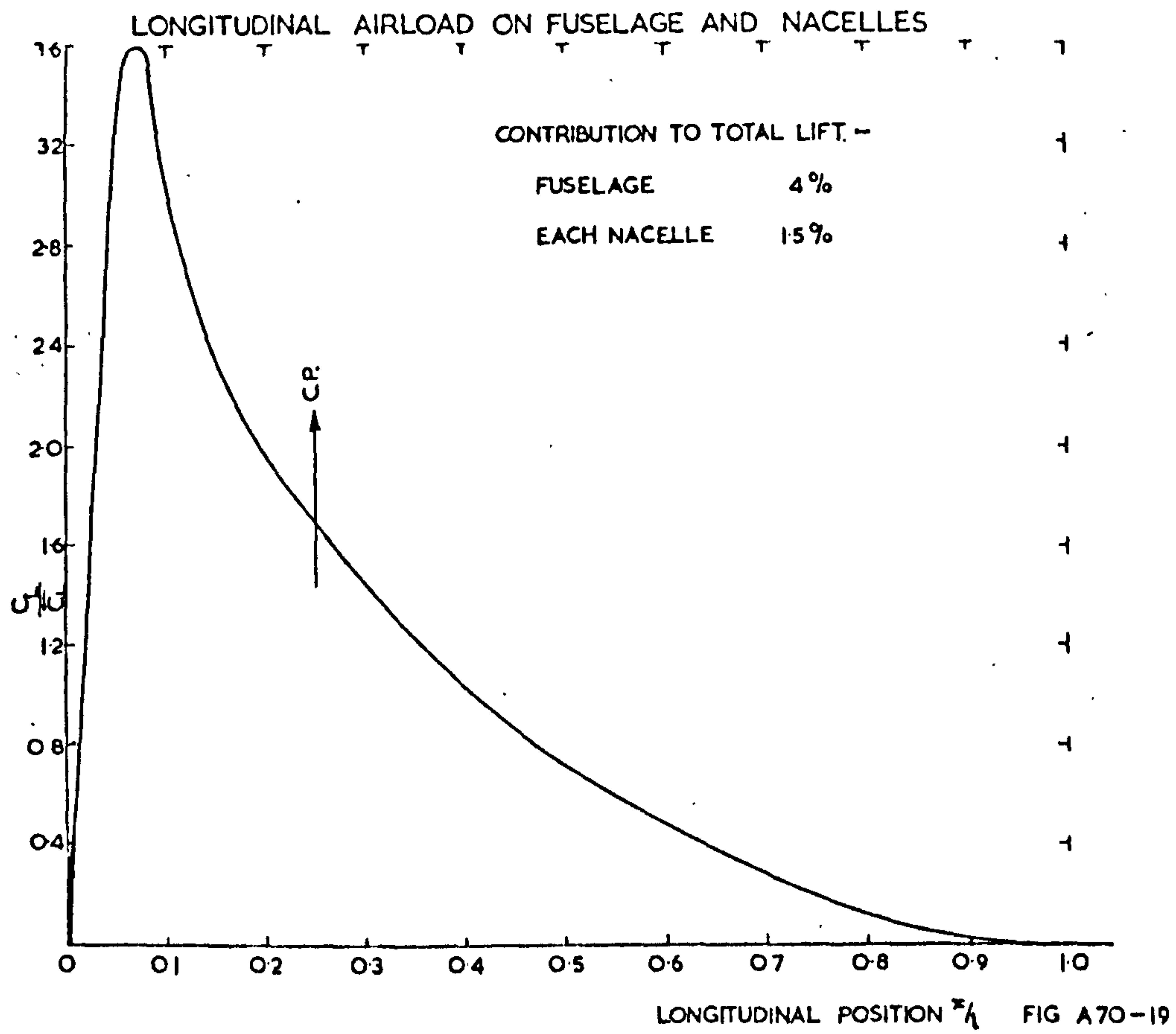


FIN SPANWISE AIRLOADING DUE TO INCIDENCE

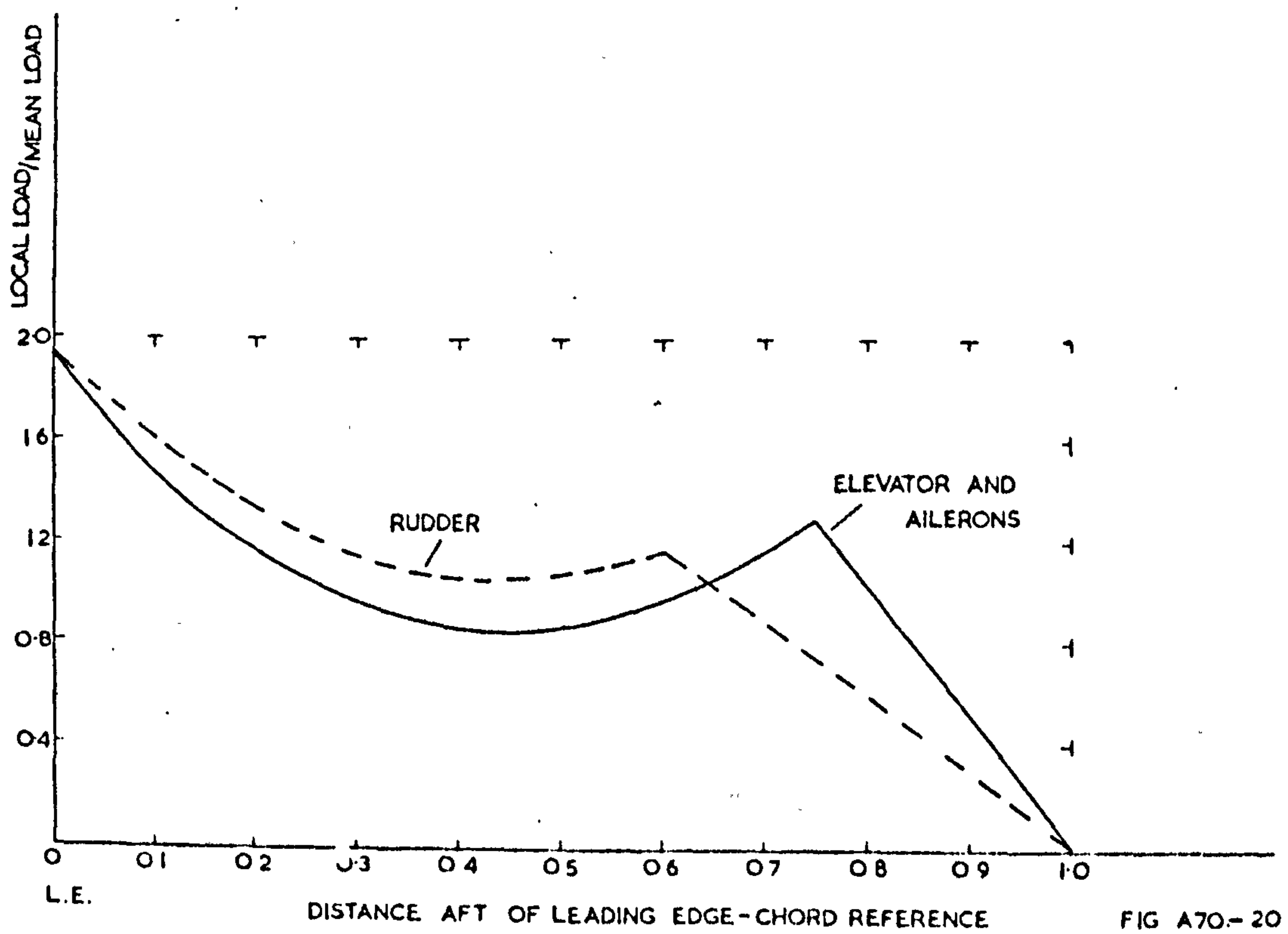


FIN SPANWISE AIRLOAD DISTRIBUTION DUE TO RUDDER

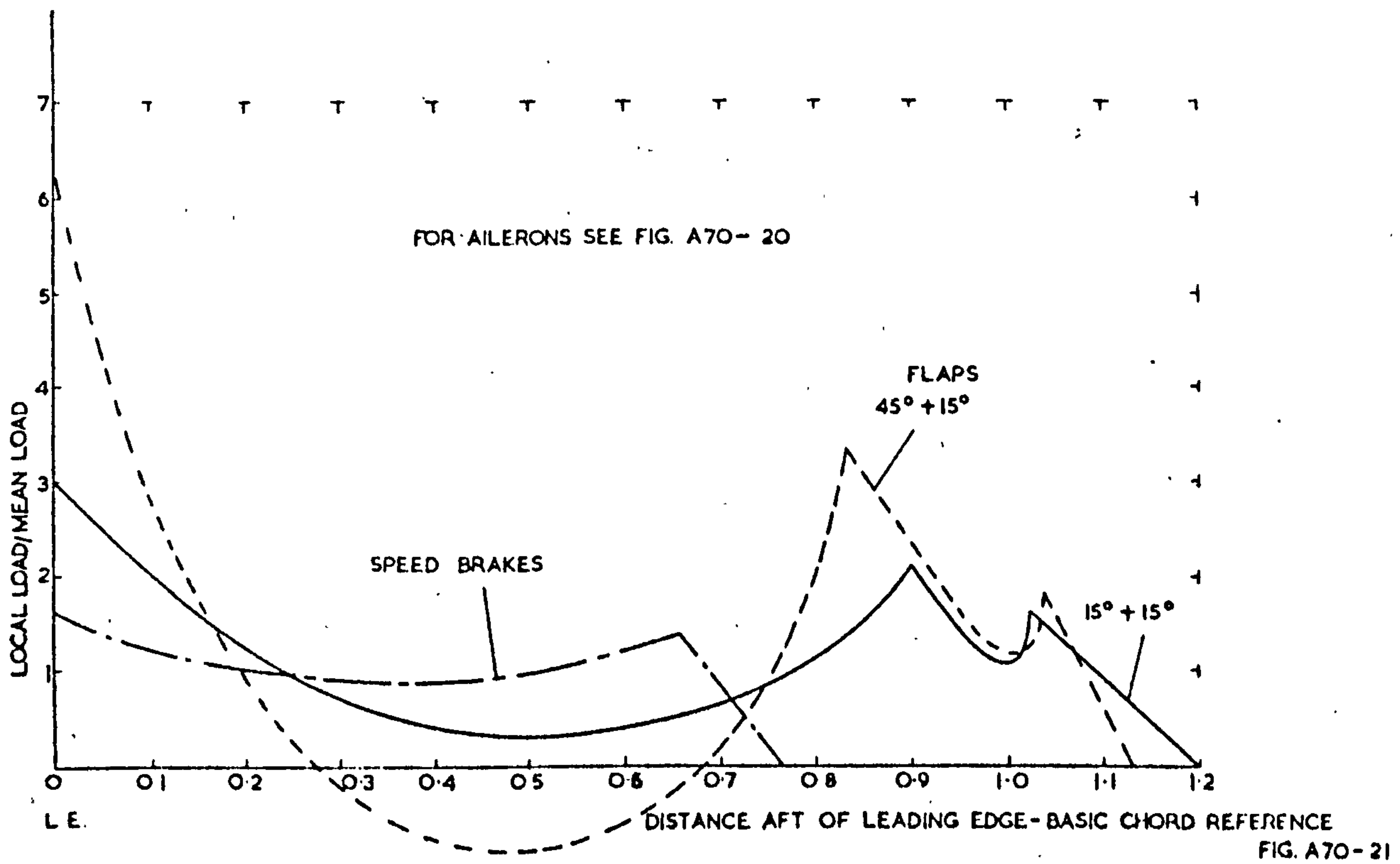




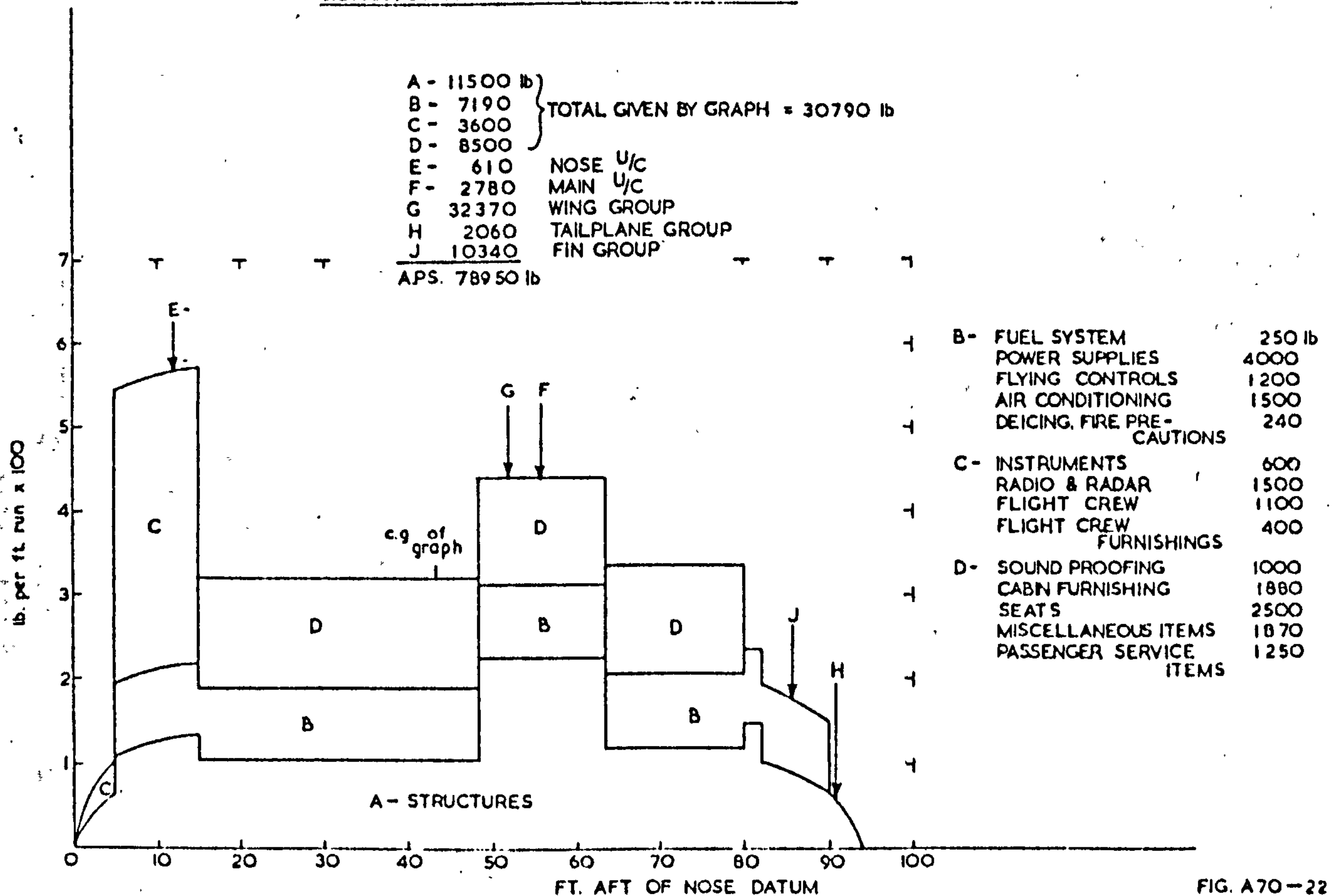
CHORDWISE LOAD DISTRIBUTION DUE TO CONTROL SURFACES



WING CHORDWISE AIRLOADING DUE TO AUXILIARY SURFACES



LONGITUDINAL INERTIA DISTRIBUTION



LIFT ENGINE NACELLE INERTIA DISTRIBUTION

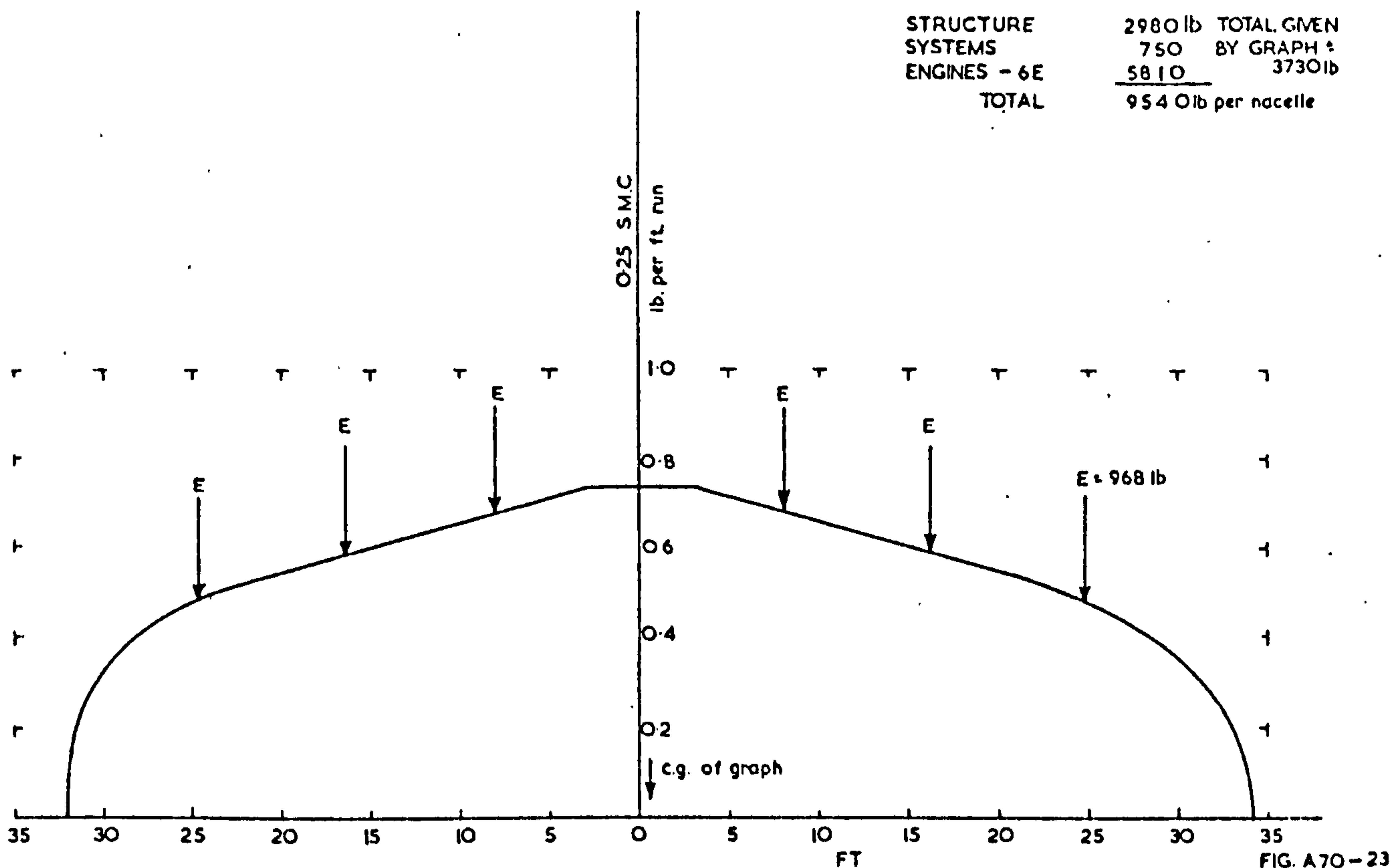
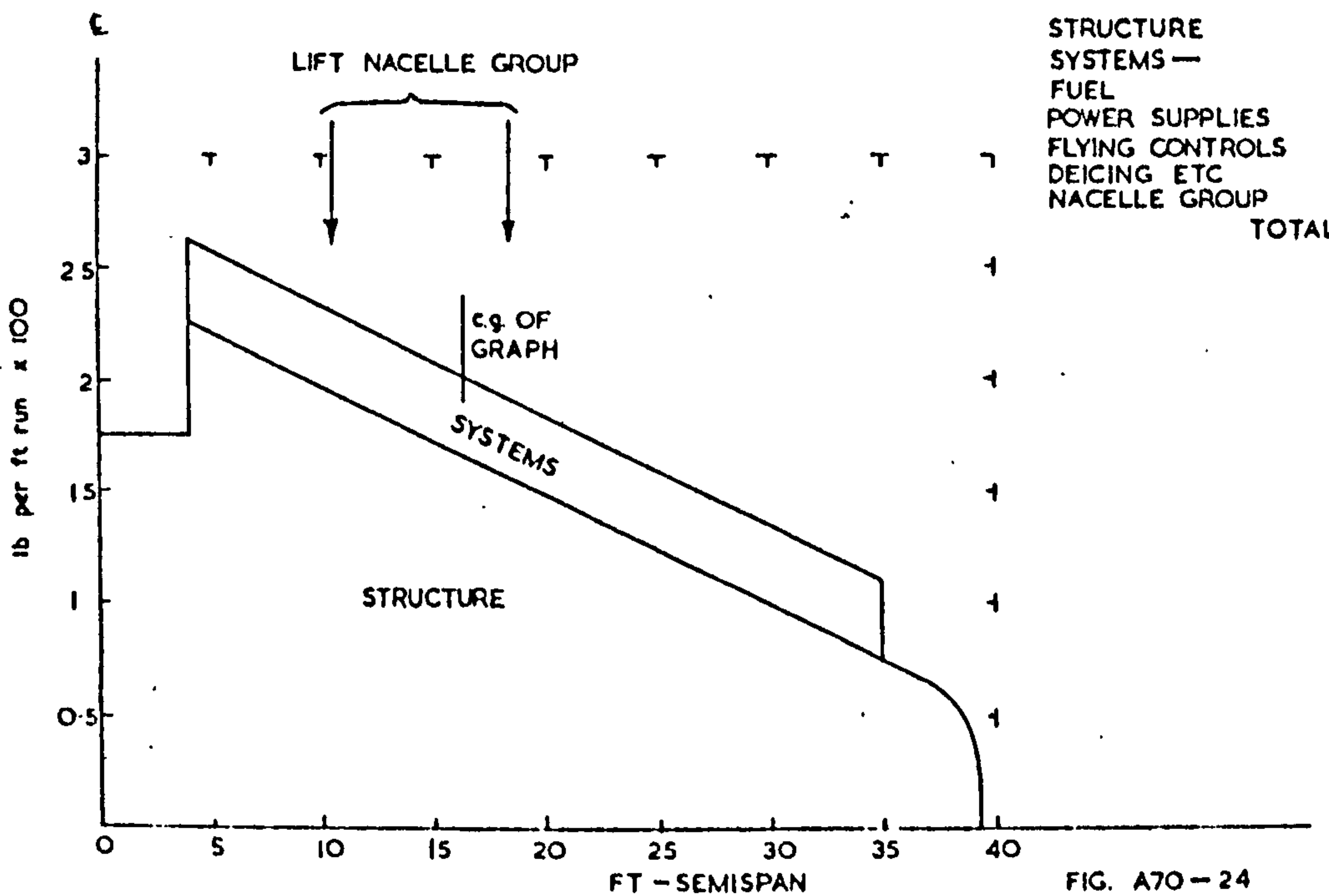


FIG. A70-23

WING SPANWISE INERTIA DISTRIBUTION

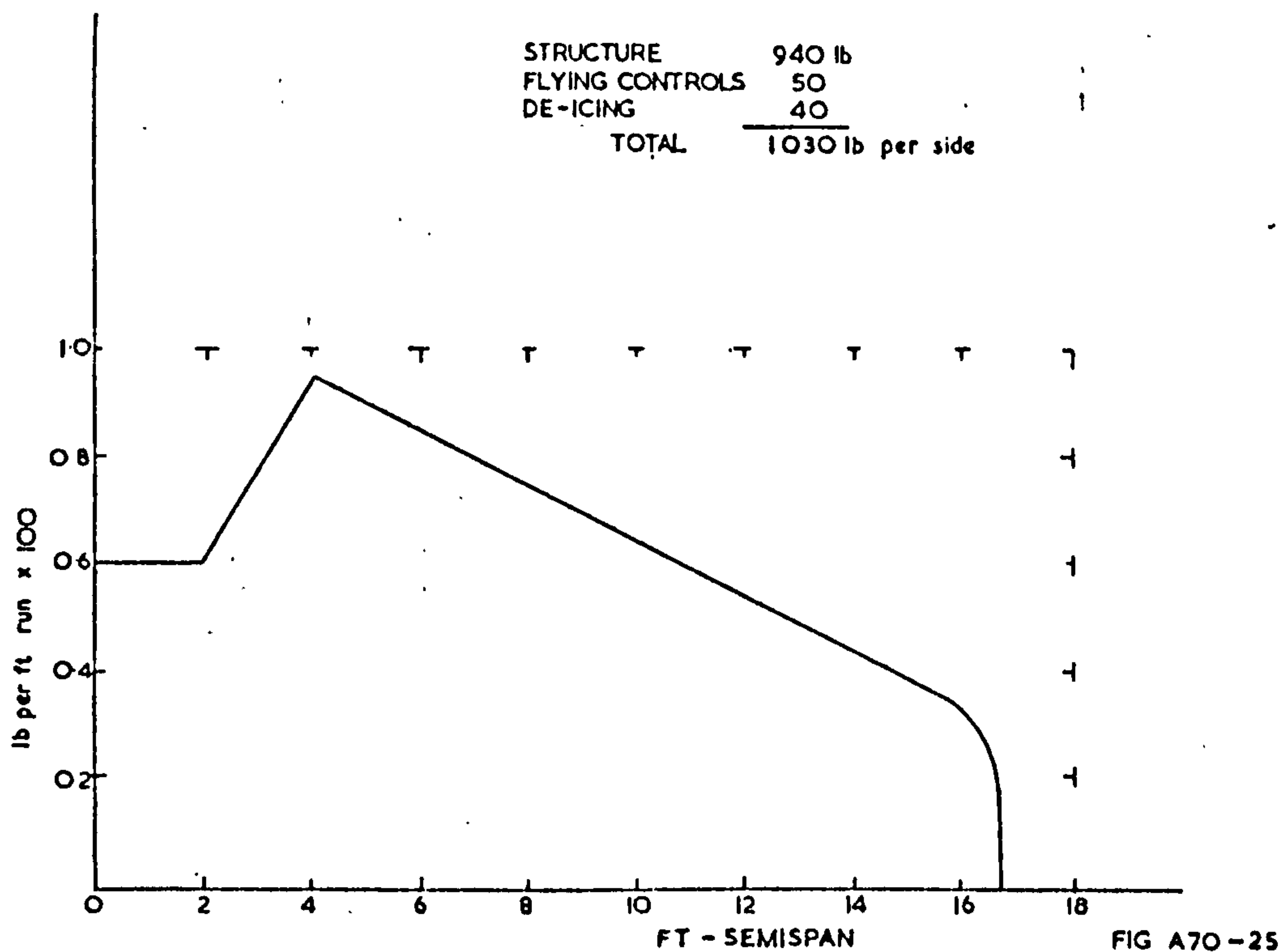


STRUCTURE
SYSTEMS —
FUEL
POWER SUPPLIES
FLYING CONTROLS
DEICING ETC
NACELLE GROUP
TOTAL

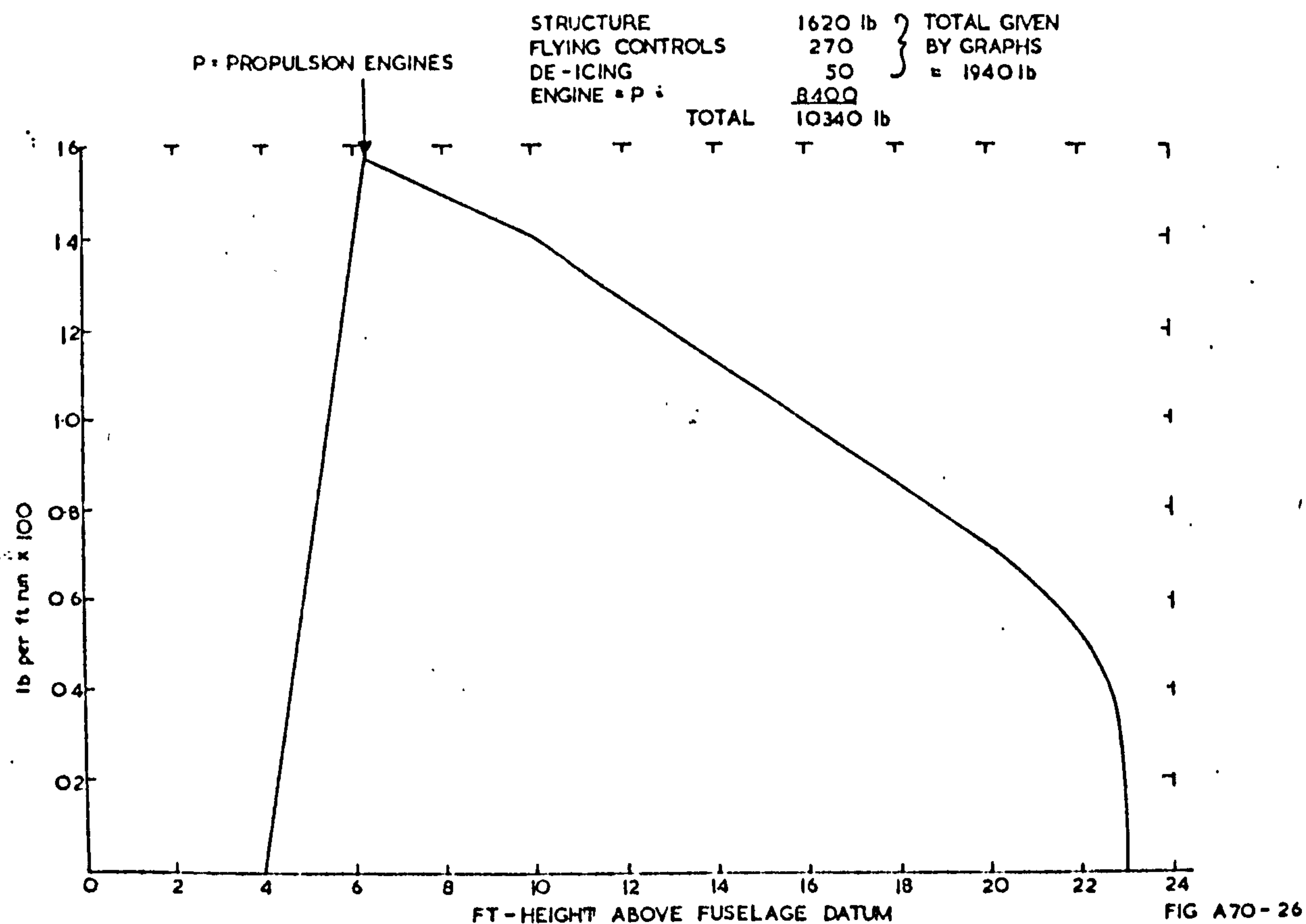
5575 lb	} TOTAL GIVEN BY GRAPH
500	
190	
150	
230	
<hr/> 2540	= 6645 lb
<hr/> 16185 lb per side	

FIG. A70-24

TAILPLANE SPANWISE INERTIA DISTRIBUTION



FIN SPANWISE INERTIA DISTRIBUTION



APPENDIX E

Data for Proposed RTOL Aircraft

(This information has been extracted from the project document, DES 7400, prepared by the author and dated August 1974)

E1. DESIGN REQUIREMENTS

The aircraft was designed to meet B.C.A.R. requirements, Section D, Category A at normal take off mass of 58,900 kg. The design value of the cruise speed V_C was 185 m/s (360 knots) EAS or $M = 0.83$ whichever was the lesser. The corresponding values of V_D were 206 m/s (400 knots) EAS and $M = 0.90$. These values are shown in Figure A74-6.

The airframe was designed to have a life of 40,000 hours with an average flight duration of 40 minutes. Typical flight profiles are shown in Figure A74-8. In addition to cruise flights allowance was made for a proportion of low level, short range positioning and training flights.

The cabin differential pressure of 0.55 bar (8 psi) ensured that cabin altitude need never exceed 2.44 km (8000 ft) and could be maintained at a lower level for many operations.

The undercarriage design vertical velocity of 5 m/s (16.4 ft/sec) catered for a steep approach condition with a partial flare.

A typical make up of different flight profiles is given in Table 3.

E2. GEOMETRY

E2.1 Wing (See Fig.A74-3)

Gross area	136 m ²
Span	35.42 m
Aspect ratio	9.22
Root chord (centreline)	6.34 m
Chord at kink (0.47 semispan)	3.645 m
Tip chord (nominal)	2.024 m
Leading edge sweepback - inboard, approx	14.5°
outboard, approx	6.0°
Sweep of 0.6c line	0°
Standard mean chord, \bar{c}	3.84 m
Aerofoil section:- root	13% thickness supercritical
tip	12% thickness supercritical
(See Figure A74-4)	
Wing body setting angle, rel. to chord line	+1°
Dihedral on 0.6c line	3.0°
Location of 0.25 \bar{c} forward of 0.6c line	1.64 m
Location of 0.6c line aft of fuselage nose	20.21 m
Location of 0.6c line, at centreline, below datum	1.55 m

E2.2 Ailerons (See Fig.A74-4)

Type:- Round nose	
Aileron chord/wing chord	0.3
Movement	±20°
Inboard end from aircraft centreline	13.12 m
Outboard end from aircraft centreline	17.71 m

E2.3 Trailing edge flaps (See Fig.A74-4)

Type:- Triple slotted NASA	
Flap chord/wing chord outboard of leading tank:-	
a) Forward segment	0.36
b) Centre segment	0.26
c) Rear segment	0.16

... continued

Take off flap setting:-		20°
(full extension forward segment as a single slotted type)		
Landing setting		20° + 20° + 20°
Inboard end from aircraft centreline		2.286 m
Outboard end from aircraft centreline		13.092 m
E2.4	<u>Leading edge slats</u> (See Fig.A74-4)	
	Slat chord/wing chord	0.11
	Inboard end from aircraft centreline	8.25 m
	Outboard end from aircraft centreline	17.20 m
E2.5	<u>Spoilers</u> (See Fig.A74-4)	
	Spoiler chord/wing chord	0.10
	Maximum movement	20°
	Inboard end from aircraft centreline	2.286 m
	Outboard end from aircraft centreline	13.092 m
	Leading edge of spoiler from wing leading edge (outboard of kink)	0.64 c
E2.6	<u>Tailplane</u> (See Fig.A74-3)	
	Gross area	33.58 m ²
	Span	14.2 m
	Aspect Ratio	6.0
	Root chord (centreline)	2.98 m
	Tip chord (nominal)	1.75 m
	Sweepback of leading edge, approx.	6.0°
	Sweep of 0.25 c line	0°
	Aerofoil section:- 12% thickness supercritical (see Fig. A74-4)	
	Dihedral	0°
	Movement	nose up nose down
	Location of 0.6 c line aft of fuselage nose	31.66 m
	Vertical location above fuselage datum	0.28 m

E2.7	<u>Elevator</u> (See Fig.A74-4)	
	Type:-Round nose	
	Elevator chord/tailplane chord	0.4
	Movement	$\pm 15^{\circ}$
E2.8	<u>Fin</u> (See Fig.A74-3)	
	Nominal area above fuselage datum	23.28 m ²
	Net area, above fuselage	18.37 m ²
	Height above datum	6.65 m
	Nominal height above fuselage	5.87 m
	Aspect Ratio, based on nominal area	1.90
	Aspect Ratio, based on net area	1.88
	Root chord, on fuselage datum	5.00 m
	Tip chord (nominal)	2.00 m
	Sweepback of leading edge, approx.	35.0°
	Aerofoil section :- 12% thickness symmetrical (see Fig.A74-4)	
	Distance of intersection of leading edge with fuselage datum, aft of fuselage nose	27.41 m
E2.9	<u>Rudder</u> (See Fig.A74-4)	
	Type: Round nose	
	Rudder chord/fin chord	0.4
	Height of rudder root at trailing edge, above datum	1.40 m
	Height of rudder tip at trailing edge, above datum	5.035 m
	Movement	$\pm 20^{\circ}$
E2.10	<u>Fuselage</u> (See Fig.A74-2)	
	Overall length	33.50 m
	Maximum diameter	3.81 m
	Maximum internal width of cabin	3.50 m
	Cabin height	2.00 m
	Overall length of cabin	25.62 m

E2. 11 Undercarriage (See Fig.A74-2)

Type:- Nosewheel

Wheelbase, to centre of main unit bogie, 16.80 m

Track, to centre of main unit bogie 6.25 m

Main undercarriage

4 wheel bogie, sideways retracting, lever suspension type.

Tyres 36 in dia x 11 in width, 14 ply rating - 14 in rim

Tyre pressure 8.27 bar

Bogie wheelbase 1.22 m

Bogie track 0.86 m

Location of bogie centre aft of fuselage
nose 19.86 m

Nose undercarriage

Twin wheels, rearwards retracting

Tyres:- 36 in dia x 11 in width, 20 ply rating - 14 in rim

Tyre pressure 12.75 bar

Wheel track 0.80 m

Location of leg aft of fuselage nose 3.06 m

E3 POWER PLANTS

E3.1 Propulsion engines (See Fig.A74-5)

Type:- Rolls Royce RB410

Installation:- Two pairs on pylons alongside rear fuselage.

Bypass ratio, approx. 10

Sea level static thrust, each 65 kN

Overall length of twin pod 5.30 m

Overall width of pod 3.95 m

Overall depth of pod 2.17 m

Location of intake face aft of fuselage
nose 23.67 m

Location of pod datum above fuselage
datum 3.45 m

Location of inboard engine centreline,
outboard 2.70 m

Location of outboard engine centreline,
outboard 4.64 m

Pylon geometry (See Fig.A74-5)

Pylon thickness ratio 0.12

E3.2 Auxiliary Power Unit

Type:- Airesearch GTCP85C

Location of A.P.U. above fuselage datum 0.60 m

Location of A.P.U. front face aft of
fuselage nose 33.95 m

E4. MASSES, CENTRES OF GRAVITY AND MOMENTS OF INERTIA

Design normal take off mass 58900 kg

Design maximum landing mass 52000 kg

Minimum flying mass 40000 kg

Operating empty mass 35681 kg

Maximum payload 13600 kg

Maximum fuel load 18140 kg

Mass breakdown - see Table 1.

Centres of Gravity at O.E. mass, relative to 0.25 \bar{c}
and datum

Undercarriage retracted 0.80 m aft
0.30 m above

Undercarriage extended 0.81 m aft
0.16 m above

Centres of gravity range in flight 0.20 \bar{c} to 0.38 \bar{c}

Moments of Inertia - see Table 2.

E5. AERODYNAMIC INFORMATION

E5.1 Lift characteristics (overall, untrimmed)
(See Fig.A74-9)

Maximum lift coefficient:-

basic wing, tip stall 1.4
slatted wing, root stall 1.6
flaps at take off setting 2.2
flaps at landing setting 3.25

Wing no lift angle, relative to wing chord datum:-

basic	-2°
flaps at landing setting	-16.9°
flaps at take off setting	-7.7°

Slope of wing-body lift curve, lowspeed:-

basic	5.4
flaps deployed	6.5

Slope of wing-body lift curve variation with M,
a₁ - see Fig.A74-10

E5. 2 Drag characteristics

Drag polar:-

Typical cruise condition

$$C_D = 0.0174 + 0.0437C_L^2$$

Take off at sea level, undercarriage and flaps extended

$$C_D = 0.063 + 0.0437C_L^2 + 0.021\Delta C_L^2$$

Landing at sea level, undercarriage and flaps extended

$$C_D = 0.203 + 0.0437C_L^2 + 0.021\Delta C_L^2$$

where ΔC_L is increment in C_L due to flap deflection.

E5. 3 Pitching moment characteristics (low speed)

Pitching moment coefficient at zero lift

Wing alone, C_{M_0} , -0.07

Increment due to body and
nacelles, ΔC_M -0.01

Pitching moment increment due to flaps:-

Take off setting, ΔC_M -0.30

Landing setting, ΔC_M -0.78

Location of overall wing-body aero centre
from fuselage nose, clean 18.44 m

(Forward shift due to basic
fuselage 0.19c)

(Aft shift due to engine
nacelles 0.155c)

Spanwise variation of basic wing aero centre - see Fig.
A74-13.

E8.4 Control and stabiliser characteristics

Location of mean tailplane aero centre aft of fuselage nose	30.78 m
Spanwise variation of tailplane aero centre - see Fig. A74-13	
Location of mean fin aero centre aft of fuselage nose	30.21
Height variation of fin aero centre - see Fig. A74-14	
Rolling moment coefficient due to aileron, l_{ξ} - see Fig. A74-11	
Yawing moment coefficient due to aileron, n_{ξ}	$0.14C_{LW}$
(where C_{LW} is basic wing lift coefficient)	
Aileron hinge moment coefficient due to wing incidence, b_1	-0.22
Aileron hinge moment coefficient due to aileron angle, b_2	-0.65
Slope of tailplane lift curve variation with M, a_{1T} , - see Fig. A74-10	
Ratio of elevator lift curve slope, a_{2T}/a_{1T}	0.65
Elevator hinge moment coefficient due to tailplane angle, b_{1T}	-0.36
Elevator hinge moment coefficient due to elevator angle, b_{2T}	-0.56
Slope of fin lift curve variation with M, a_{1F} , - see Fig. A74-10	
Ratio of rudder lift curve slope, a_{2F}/a_{1F}	0.46
Rudder hinge moment coefficient due to fin angle, b_{1F}	-0.14
Rudder hinge moment coefficient due to rudder angle, b_{2F}	-0.42
Yawing moment coefficient due to rudder, n_{ζ} , approx.	$-0.029a_{1F}$
Rolling moment coefficient due to rudder, l_{ζ}	$-0.014C_{LW} + 0.023$
Slope of lift decrement due to operation of spoilers (Both wings 20° limit)	-0.30
Slope of drag increment due to operation of spoilers (Both wings 20° limit)	0.08

E5. 5 Stability characteristics

Downwash at tailplane - see Fig.A74-12

Rolling moment coefficient due to:-

Rolling moment, ℓ_p - see Fig.A74-11

Sideslip, ℓ_v -0.145

Yawing, ℓ_r $0.017 + 0.21C_{LW}$

Yawing moment coefficient due to:-

Rolling, n_p -0.07 C_{LW}

Sideslip, n_v , overall $+a_{1F}(0.057 + 0.026\bar{x}) - 0.054$
tail off -0.054

(where \bar{x} is position of c.g. as a fraction of \bar{c})

Yawing, n_r $-\left[0.002 + a_{1F}(0.037 + 0.026\bar{x})\right]$

Sideforce coefficient due to:-

Sideslip, y_v -0.33

Tailplane rolling moment coefficient
due to sideslip, K_β 0.12

(Note all derivatives are based on the reference areas and dimensions quoted in paragraph 3. Hinge moment coefficients are based on control surface chord and area aft of the hinge line. All angular measurements are in radians unless otherwise stated)

E6. LOAD DISTRIBUTIONS

E6 .1 Aerodynamic loads

The wing spanwise load distributions due to incidence, flap, aileron, spoiler and slat deflections are shown in Figures A74-15 and 16. The tailplane spanwise load distribution due to both incidence and elevator deflection is given in Figure A74-17 whilst the corresponding information for the fin and rudder is to be found in Figure A74-18. Figure A74-24 shows a typical lift distribution along the fuselage. The shape of this distribution is dependent upon incidence and the diagram given is a mean for initial loading calculations.

Chordwise load distributions vary substantially with Mach number and lift coefficient. The curves given should only be used for local design of the various components and not for overall balance calculations. Typical wing chordwise loading due to incidence is shown in Figure A74-19 whilst distributions due to flaps, control surface, spoiler and slat deflections are given in Figures A74-20 to A74-23. The chordwise loading on the tailplane and elevator may be taken as similar to the wing whilst that due to the rudder is covered by Figure A74-23. Chordwise loading due to fin incidence may be evaluated from DES 545.

E6.2 Inertia loads

The inertia load distributions are presented appropriate to the operating empty mass of the aircraft. It is necessary to add appropriate increments for the payload and fuel according to the particular mass and centre of gravity condition being investigated. The basic longitudinal and spanwise inertia distributions are given in Figures A74-25 to A74-28 for the fuselage, wing, fin and tailplane respectively.

It is essential to note that each of these inertia distributions represents only one of the two orthogonal components which make up the moments of inertia quoted in Table 2. In order to obtain a correct overall balance it is essential to compensate for this effect. A simple way of doing this is to factor the graphical load distributions accordingly. For the operating empty weight case the factors are:-

Pitching, overall longitudinal distributions, factor to cover the z arms	1.065
Rolling, overall spanwise distributions, factor to cover the z arms	1.257
Yawing, summate the longitudinal and spanwise distributions. (The spanwise effect is 20.2% of the total)	

TABLE E1

Mass Breakdown

<u>Component</u>		<u>Mass kg.</u>	<u>%</u>
Wings	6484		11.0
Fuselage	5783		9.8
Tailplane	726		1.2
Fin	544		0.9
Engine Pylons	417		0.7
Main undercarriage	1828		3.1
Nose undercarriage	<u>374</u>		0.6
Total Structure		16156	27.4
Powerplant	8165		13.9
A.P.U.	272		0.5
Fuel System	<u>445</u>		0.7
Total Powerplant services		8882	15.1
Power supplies	1906		3.2
Flying controls	635		1.1
De-icing	345		0.6
Air conditioning	680		1.2
Radio and Radar	680		1.2
Instruments	<u>454</u>		0.7
Total Systems and Equipment		4700	8.0
Furnishings	4536		7.7
Crew	500		0.8
Expendable items	<u>907</u>		1.5
Operating Empty Mass		35681	60.6
Payload, maximum		13600	23.1
Fuel		<u>9619</u>	16.3
All up mass		58900	100

TABLE E2

Moments of Inertia

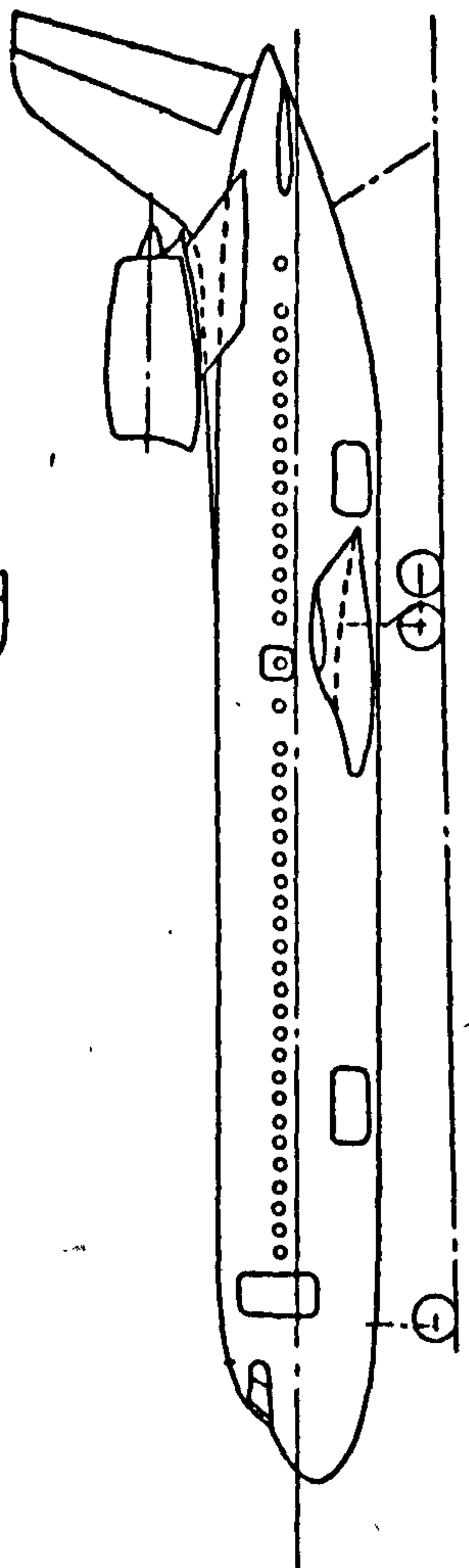
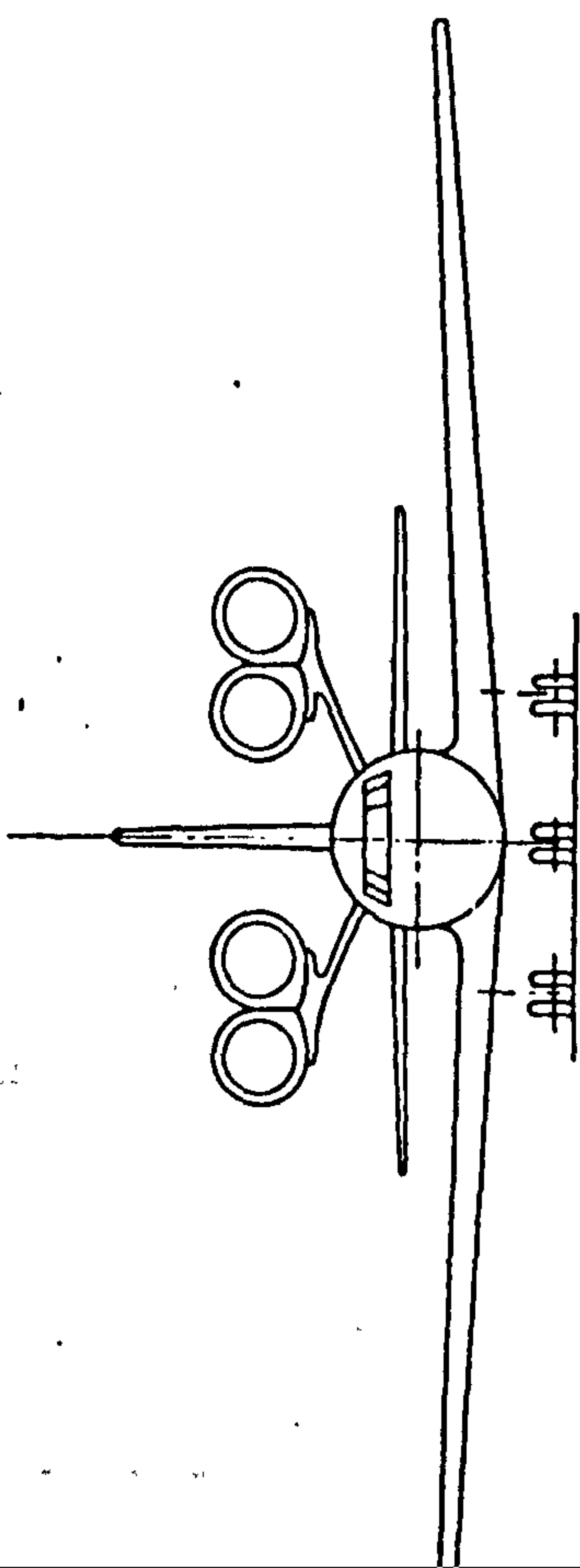
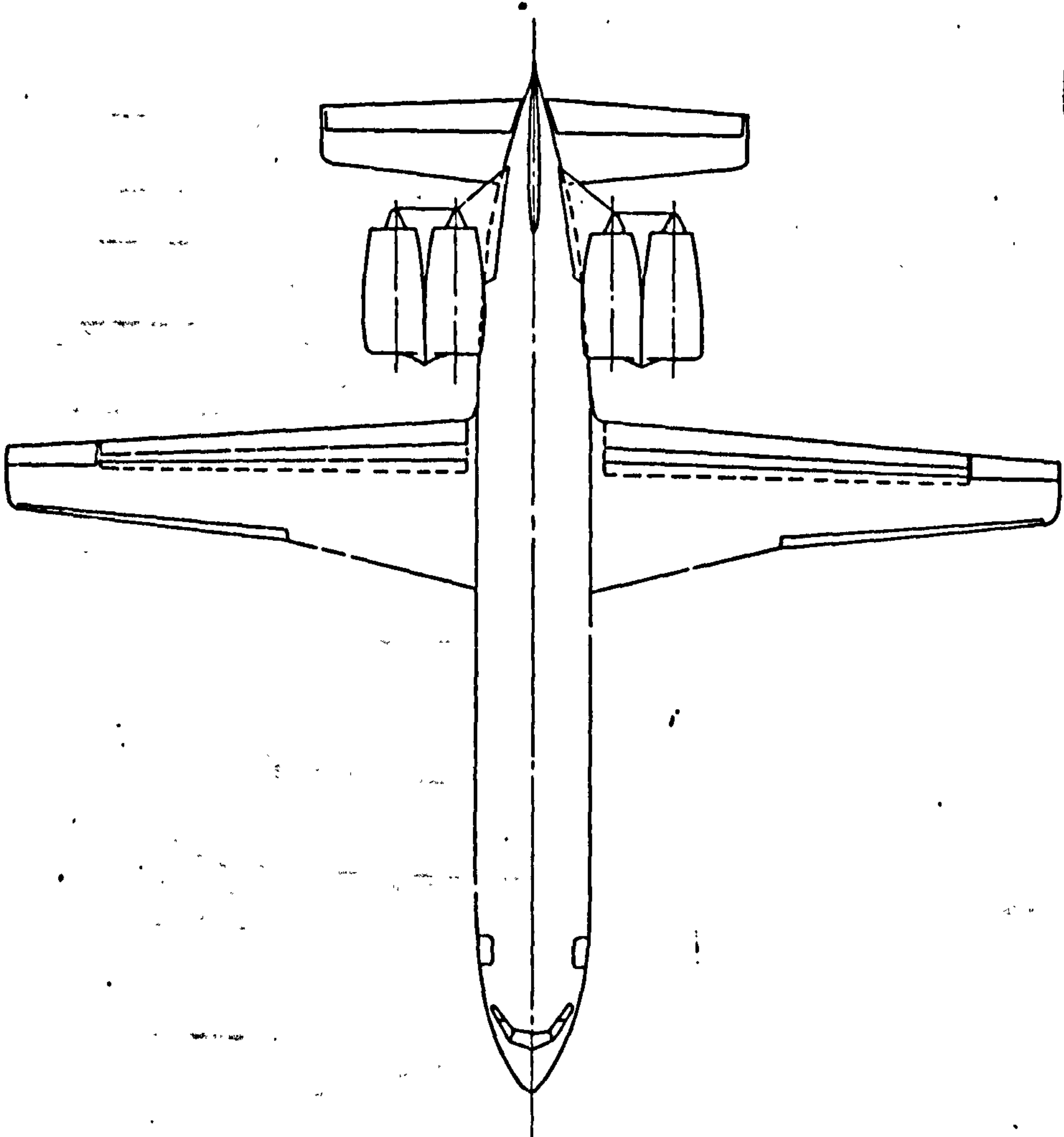
	kg m ² x 10 ⁵		
	Roll (A)	Pitch (B)	Yaw (C)
Operating Empty Mass (35681 kg)	7.25	24.2	28.5
Zero fuel, max. payload (49281 kg)	7.37	29.4	33.7
All up mass, max. payload (58900 kg)	13.1	29.8	34.1
All up mass, max. fuel load (58900 kg)	16.1	30.3	37.9

Note: The values of A, B and C at the operating empty mass condition correspond to the quoted centre of gravity and inertia distribution, Figs.A74-25 to 28. The other values quoted are typical for the conditions stated, but do vary according to the actual centre of gravity in a given case. Balance calculations must use values which are correct for a given centre of gravity, not necessarily those given above.

TABLE E3

Typical Flight Pattern

Role		Proportion of total flights in aircraft life
M	Altitude km	%
0.8	6.2	30
0.8	9.2	20
0.6	6.2	20
0.6	9.2	10
Training	0	20



DRAWN	CHE	APV	SCALE	TITLE:-	A-74
14/8/74			1/100	GENERAL ARRANGEMENT	
COLLEGE OF AERONAUTICS				DRAWING No	A 74 - I
CRANFIELD INSTITUTE OF TECHNOLOGY				SHEET	OF SHEETS
CRANFIELD					

ALL DIMENSIONS IN MM

140 SEATS - 24 ROWS AT 840 MM PITCH
146 SEATS - 25 ROWS AT 790 MM PITCH

GALLEY
TOILET

TOILET
TOILET

3810 DIA

DATUM

6250 TRACK

20210 0.6 CHORD

14/15 ROWS

10 ROWS

DATUM

1790

VENTRAL ACCESS

3350

STATIC GROUND LINE

16800 WHEEL BASE

DRAWN CHK APPR SCALE TITLE

16/8/74

1/50

COLLEGE OF AERONAUTICS
CRANFIELD INSTITUTE OF TECHNOLOGY
CRANFIELD

DRAWING No

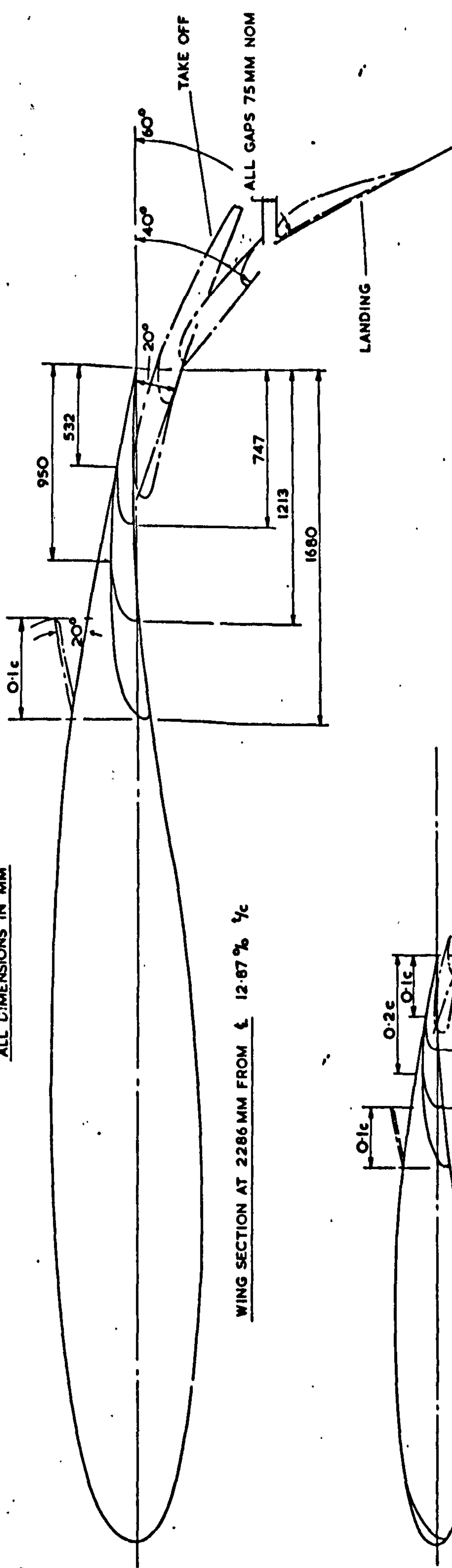
A 74-2

SHEET

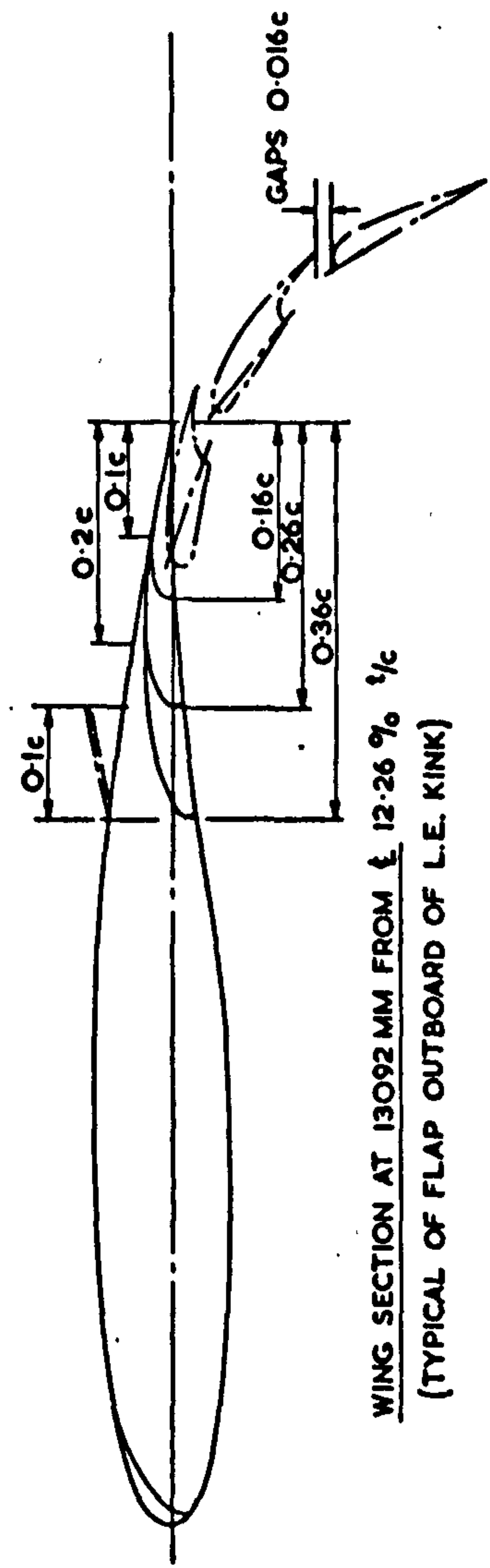
OF

SHEETS

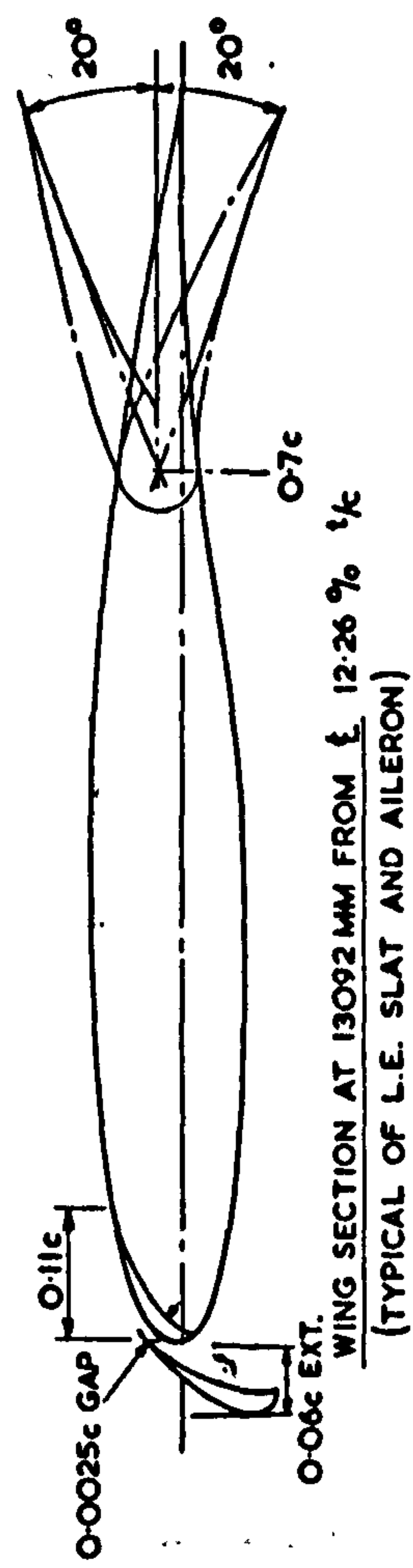
ALL DIMENSIONS IN MM



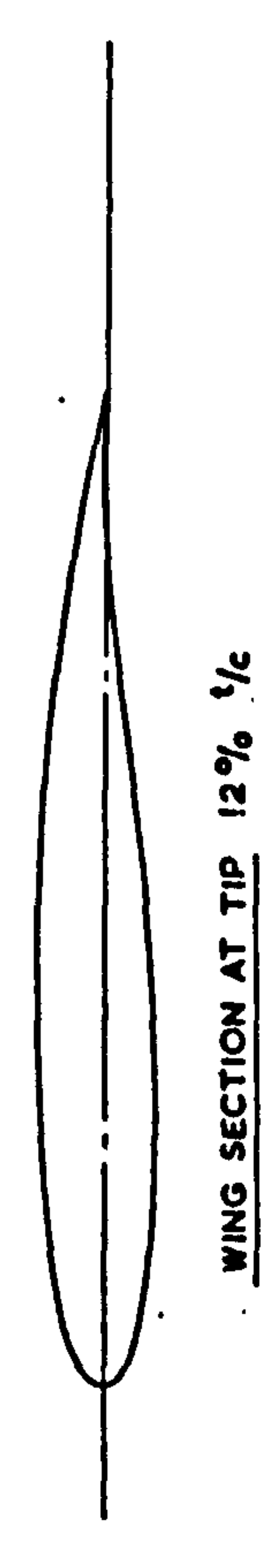
WING SECTION AT 2286 MM FROM ξ 12.87% $\frac{1}{2}c$



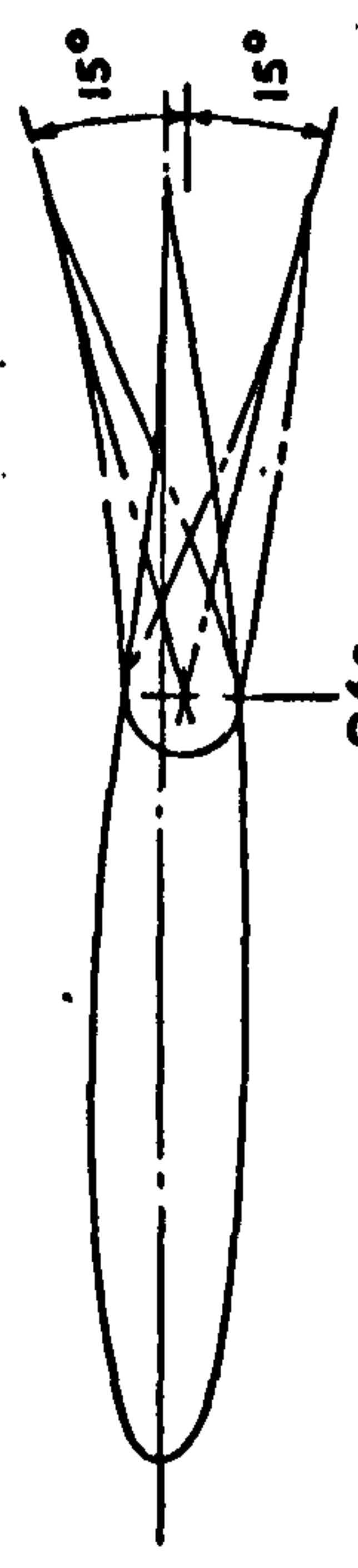
WING SECTION AT 13092 MM FROM ξ 12.26% $\frac{1}{2}c$
(TYPICAL OF FLAP OUTBOARD OF L.E. KINK)



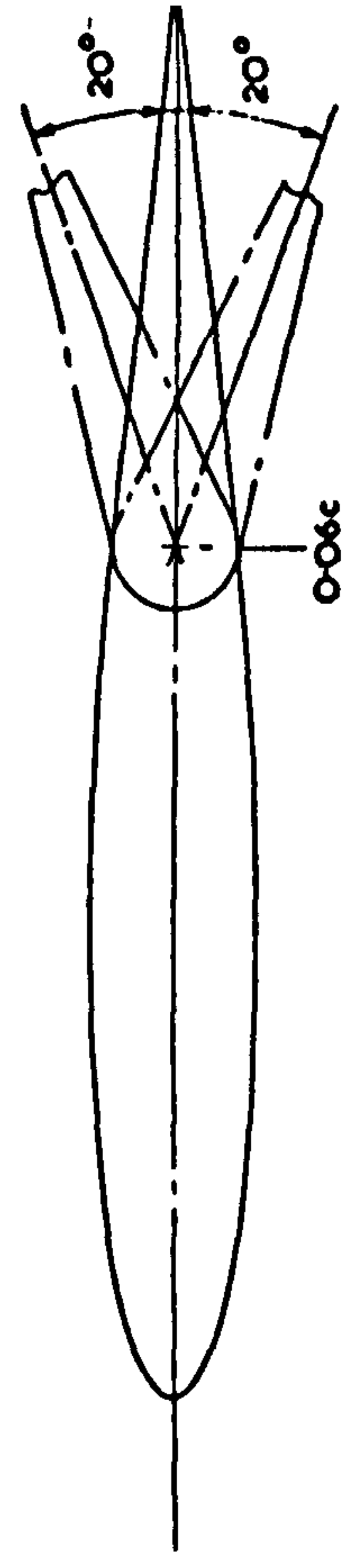
WING SECTION AT 13092 MM FROM ξ 12.26% $\frac{1}{2}c$
(TYPICAL OF L.E. SLAT ANDAILERON)



WING SECTION AT TIP 12% $\frac{1}{2}c$



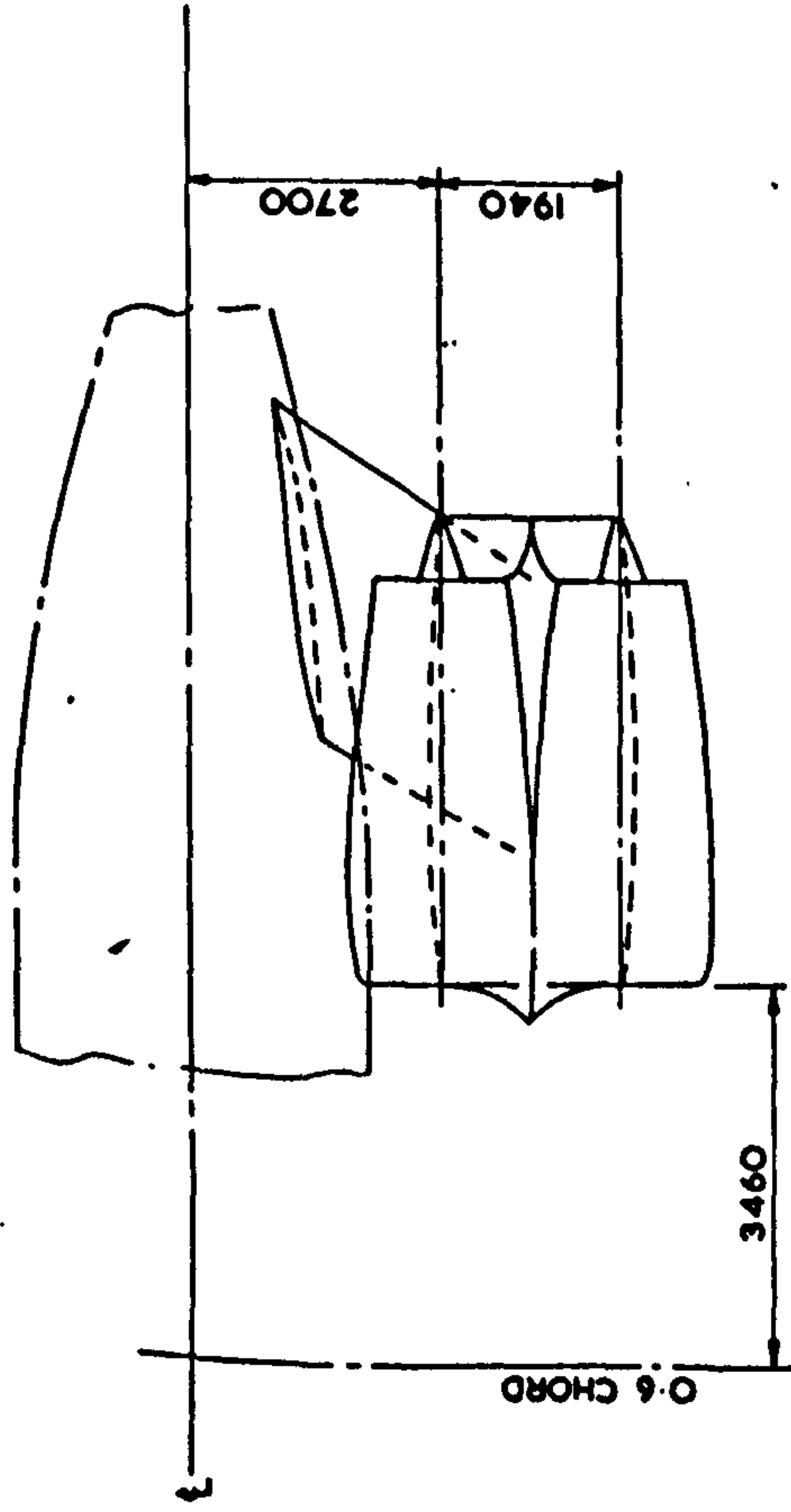
TYPICAL TAILPLANE SECTION 12% $\frac{1}{2}c$
(5518 MM FROM ξ)



TYPICAL FIN SECTION 12% $\frac{1}{2}c$
(4434 MM ABOVE DATUM)

DRAWN	CHK'D	APPV	SCALE	TITLE	A 74
11/13/74			1/10	AEROFOIL SECTIONS	
COLLEGE OF AERONAUTICS				DRAWING NO.	A 74 - 4
CRANFIELD INSTITUTE OF TECHNOLOGY				SHT	OF 3
CRANFIELD					

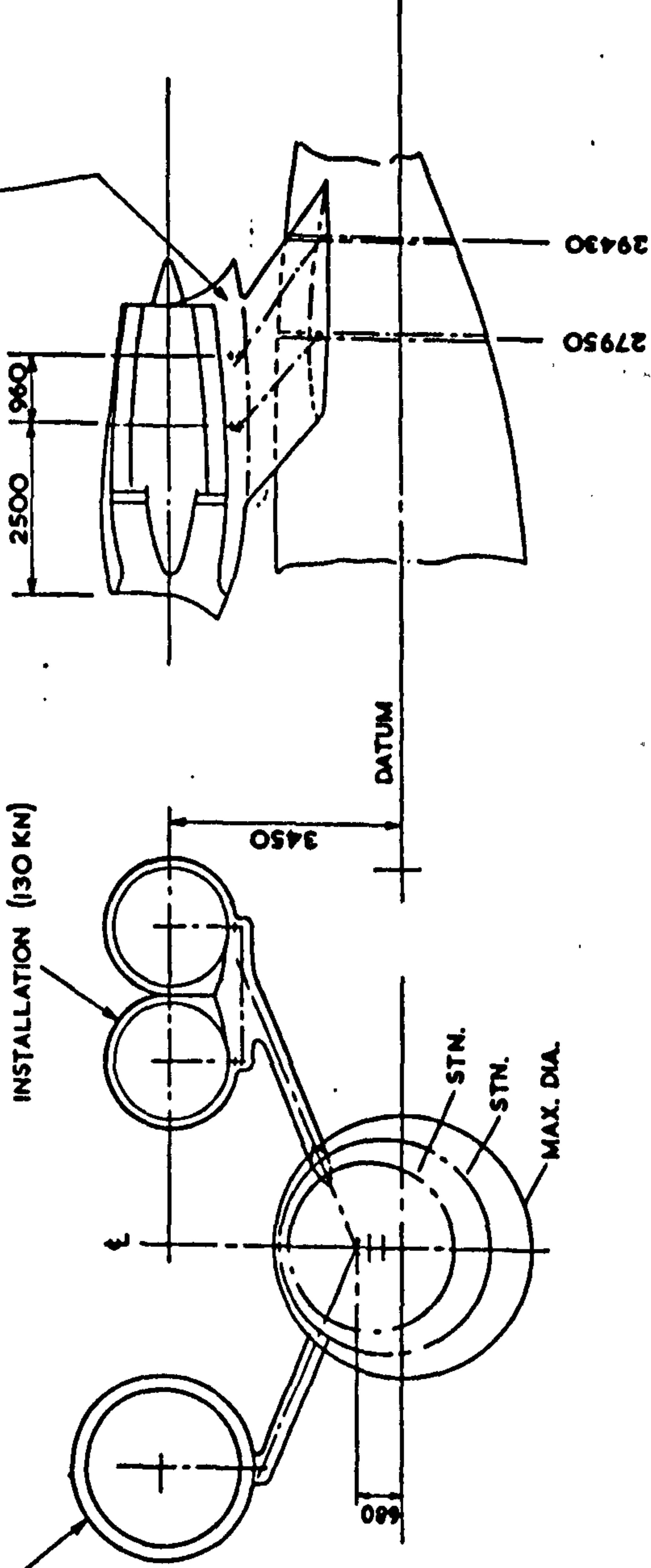
ALL DIMENSIONS IN MM.



POSSIBLE SINGLE POWERPLANT
INSTALLATION (133 KN)

TWIN RB 410 POWERPLANT
INSTALLATION (130 KN)

OIL COOLERS, AUXILIARIES
ETC. IN FAIRING



LEVEL SPEED PERFORMANCE

(MAX. CONTINUOUS RATING)

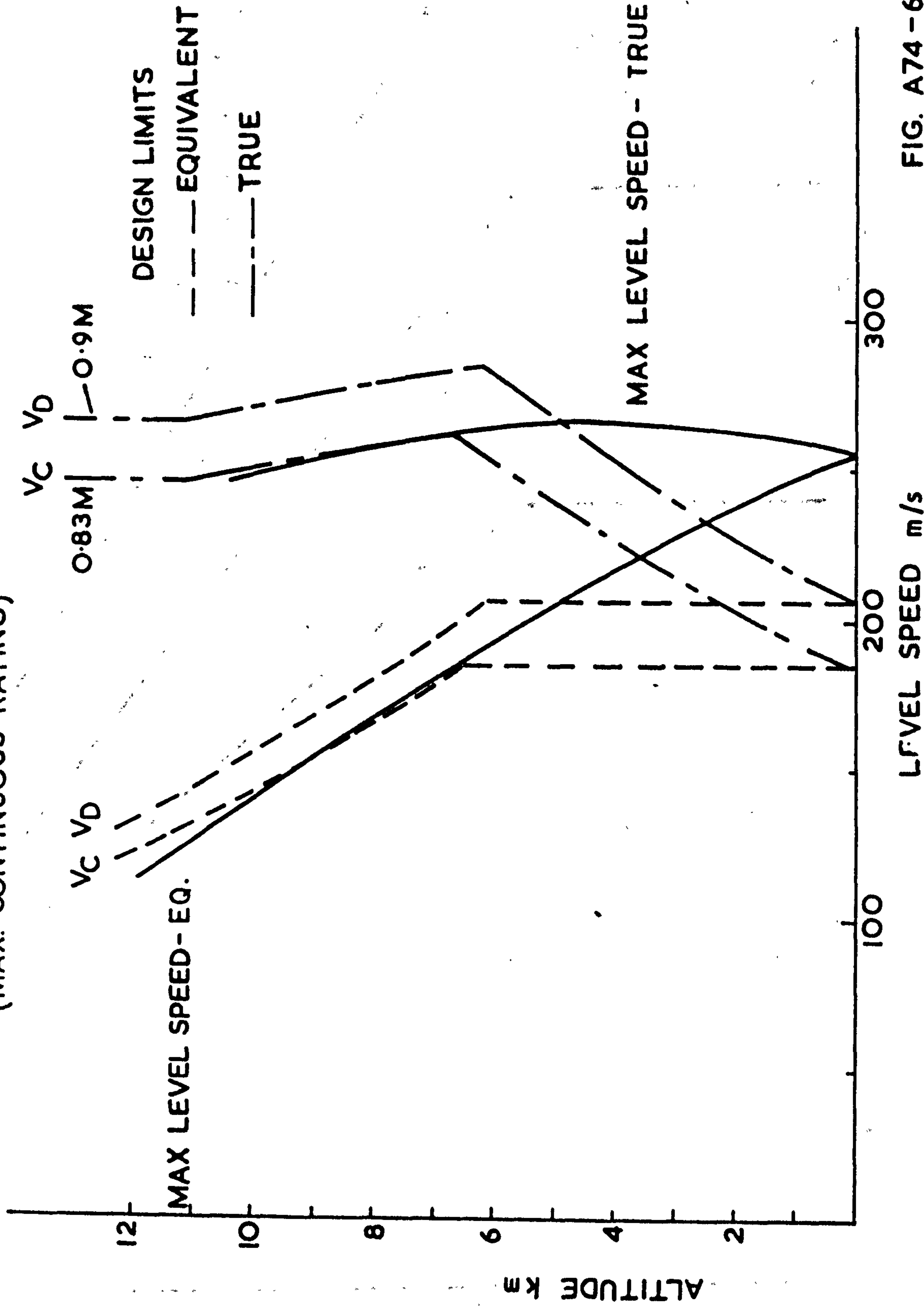


FIG. A74-6

PAYLOAD - RANGE PERFORMANCE

(LANDING RESERVE ONLY)

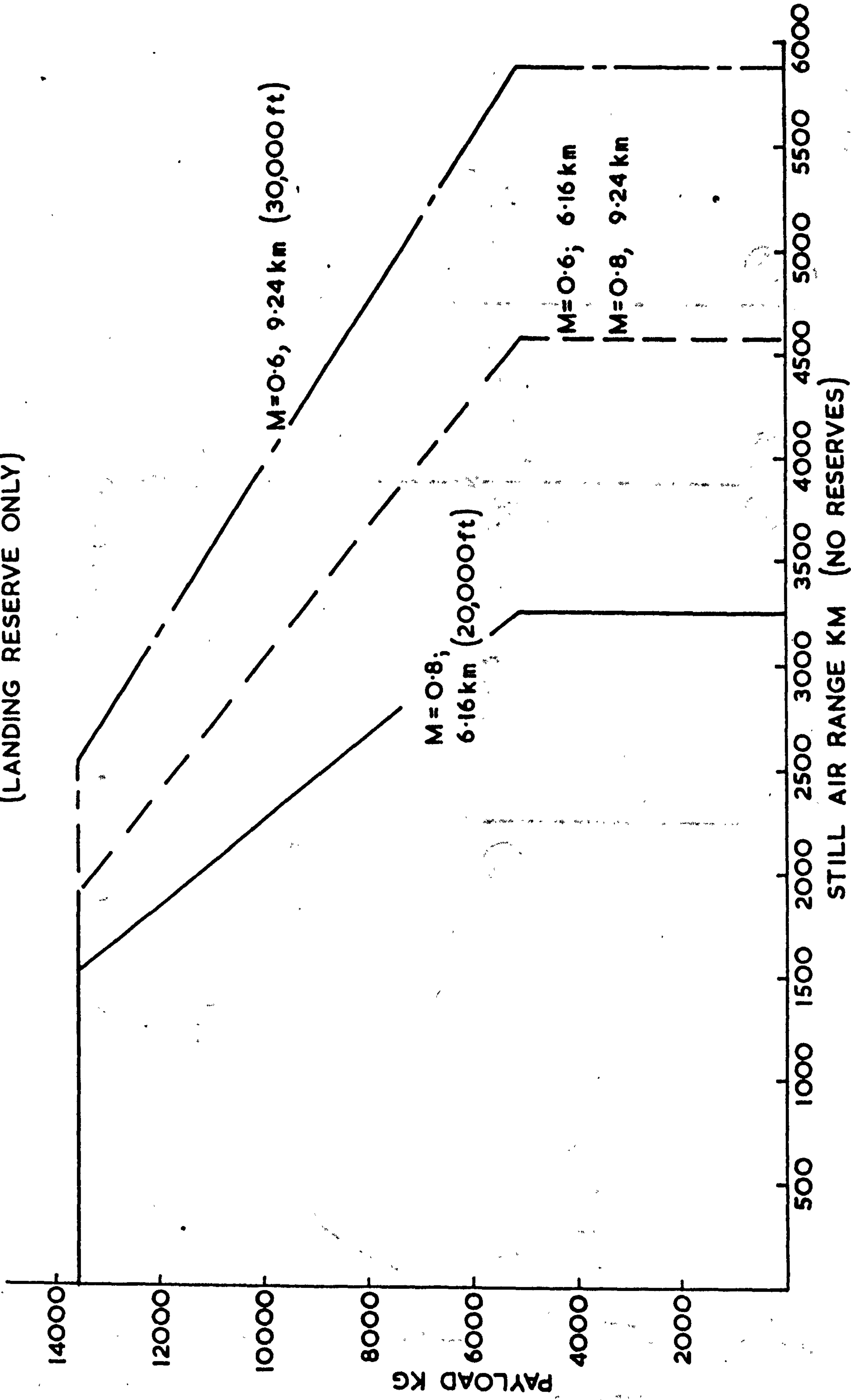


FIG A74-7

(FULL PAYLOAD)

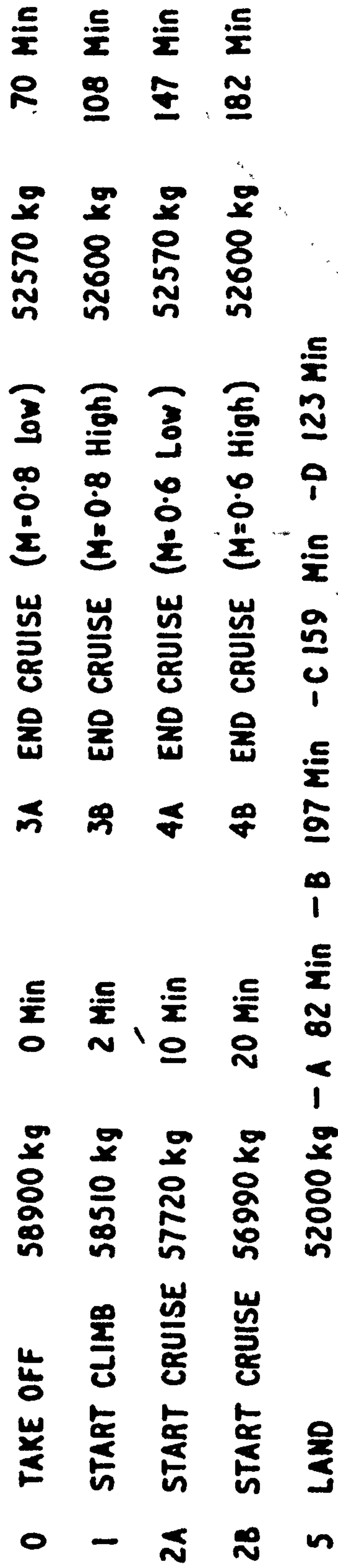
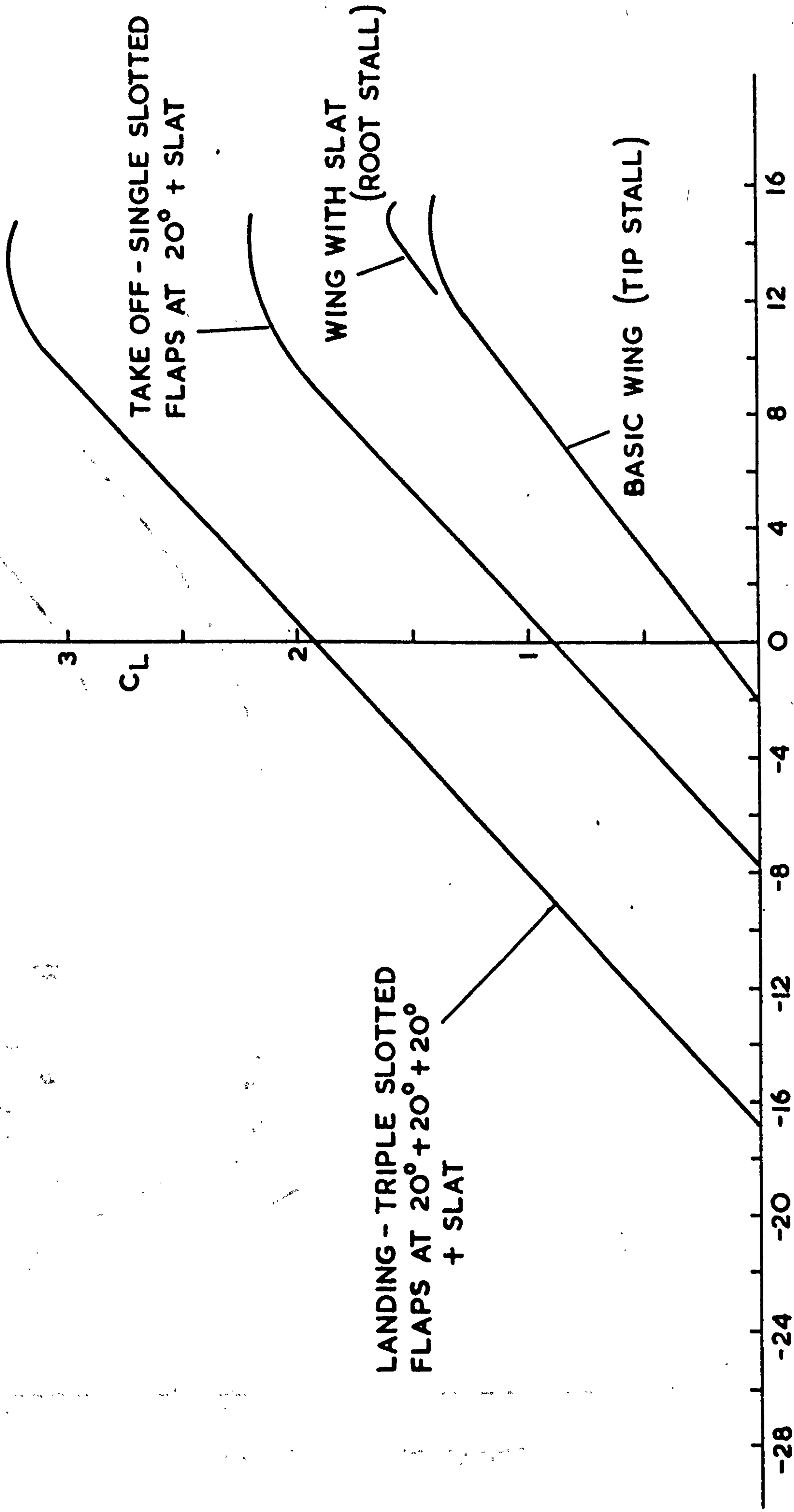


FIG A74-8

WING LOWSPEED LIFT CURVES



ANGLE OF ATTACK - RELATIVE TO CHORD DATUM - DEG

FIG A74-9

LIFT CURVE SLOPES DUE TO INCIDENCE

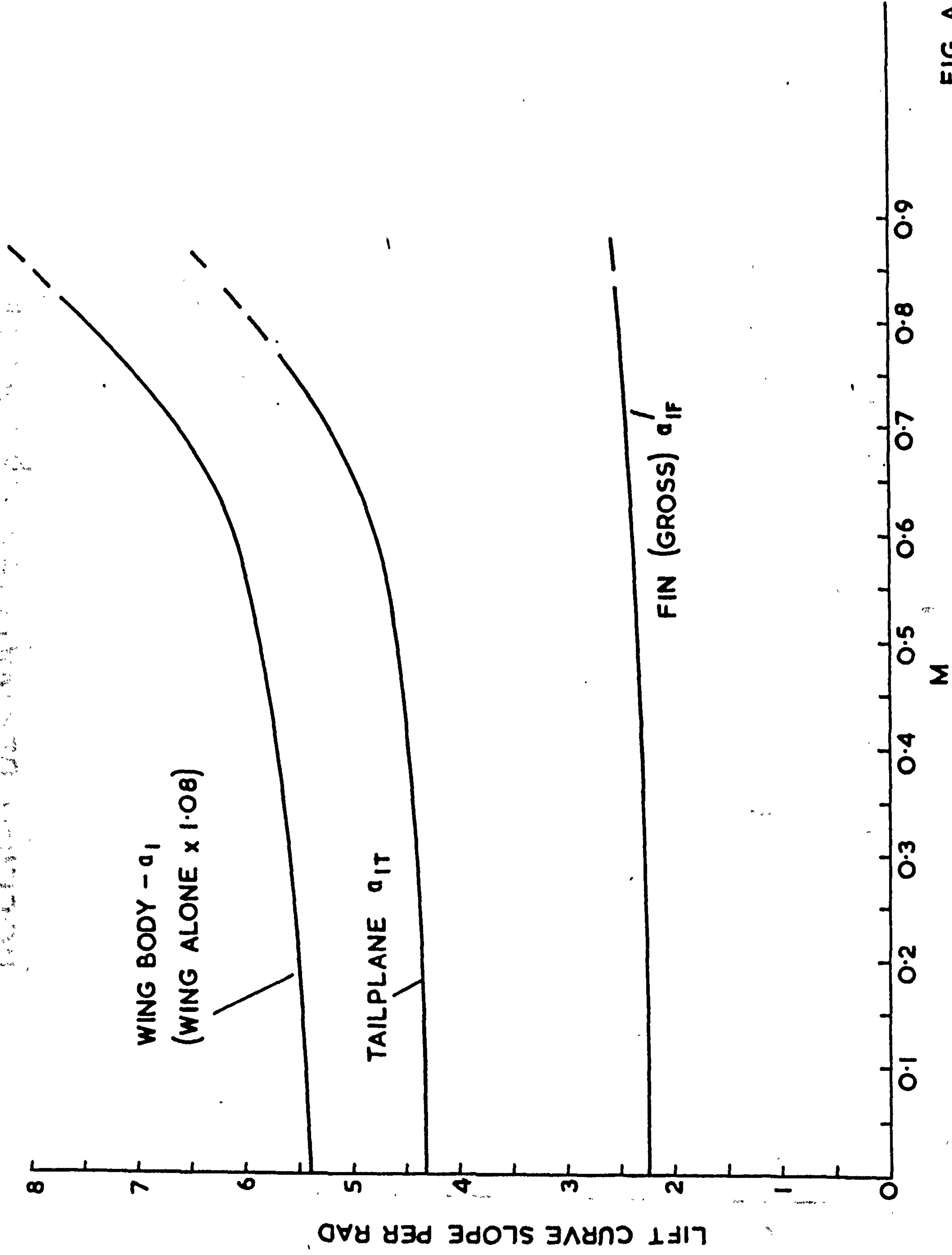


FIG A74-10

ROLLING DERIVATIVES l_p AND l_s

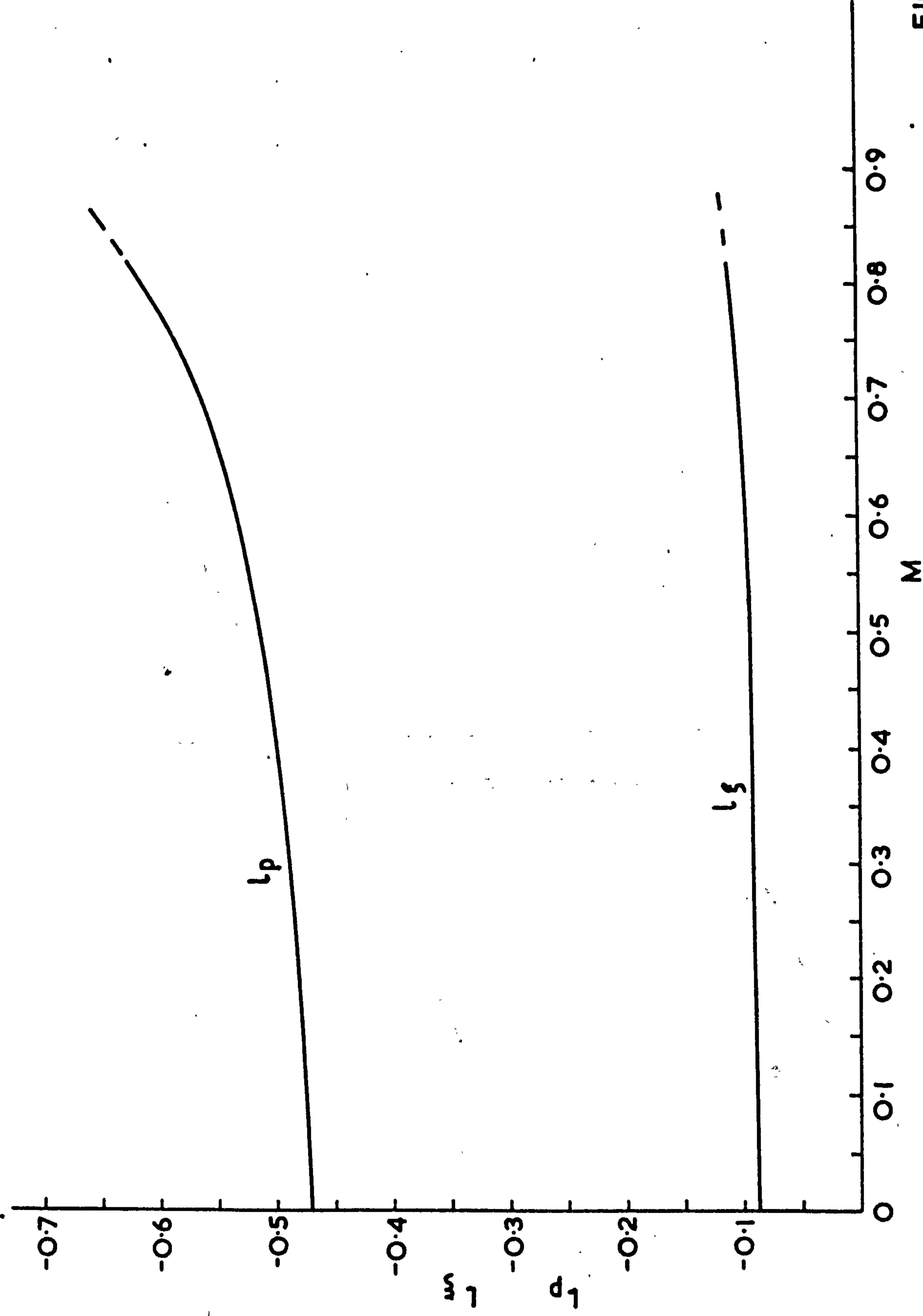


FIG. A74-II

DOWNWASH ANGLE AT TAIL PLANE

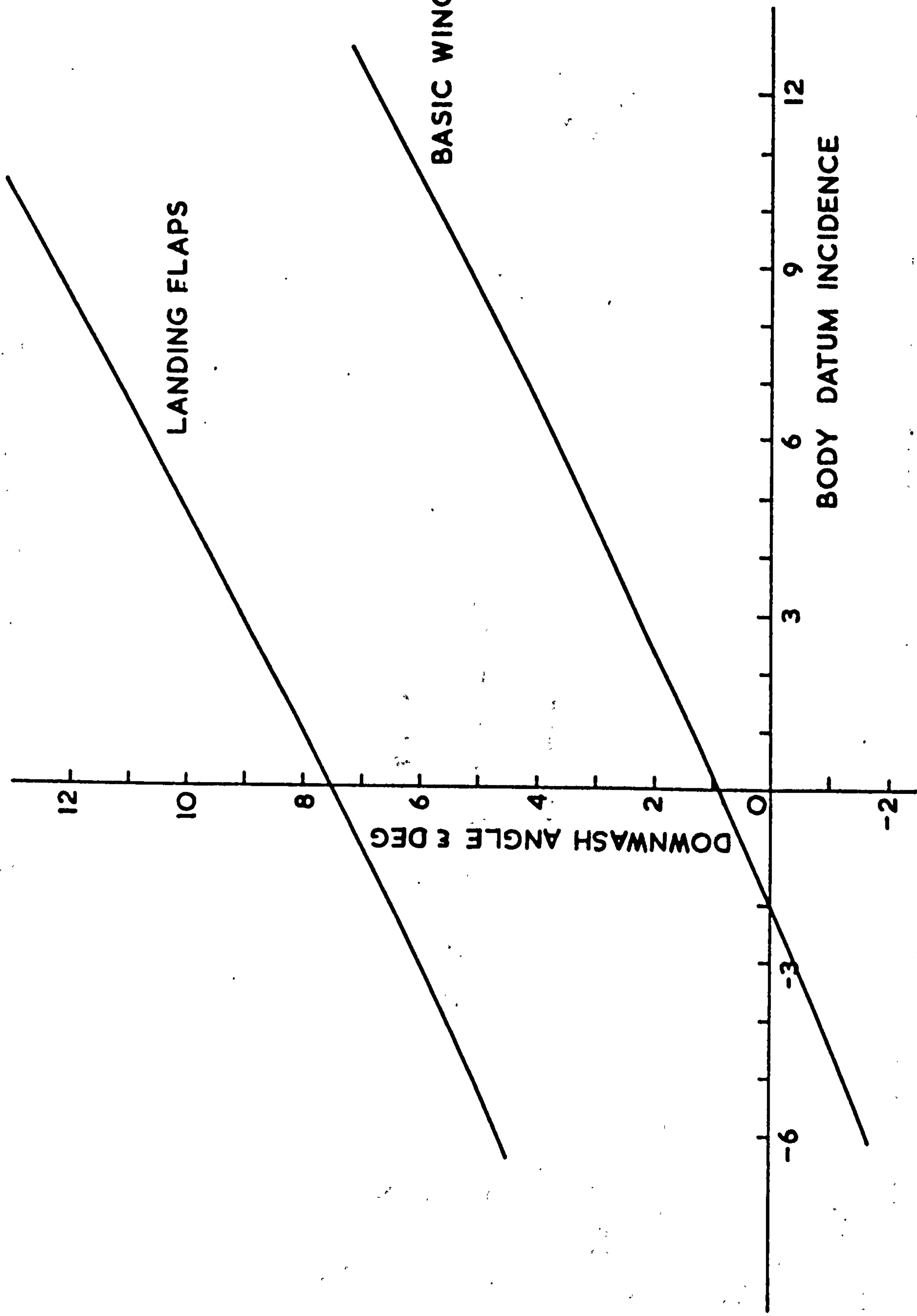


FIG. A74-12

SPANWISE VARIATION OF WING AND TAIL AERO CENTRE LOCATIONS

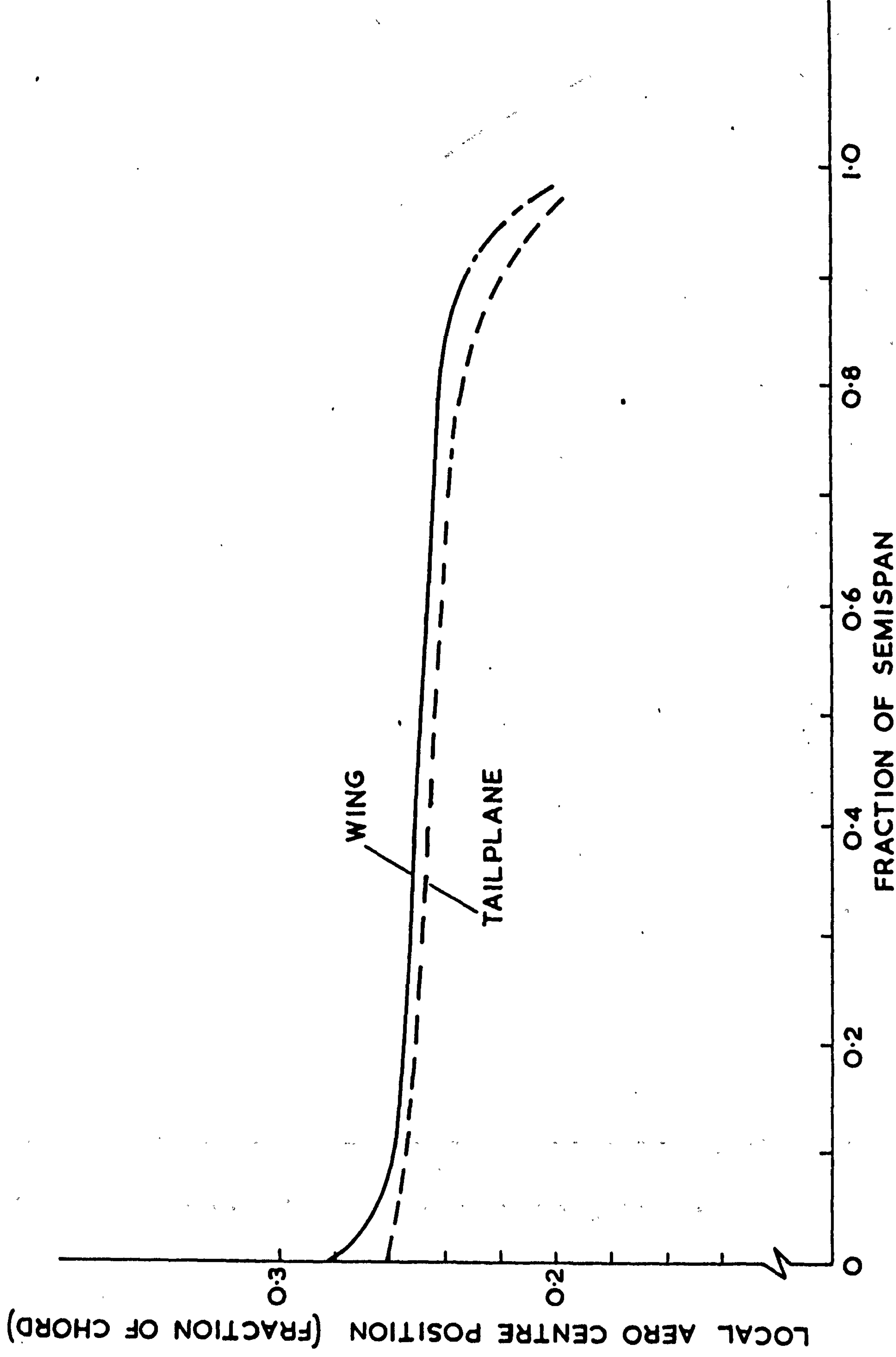
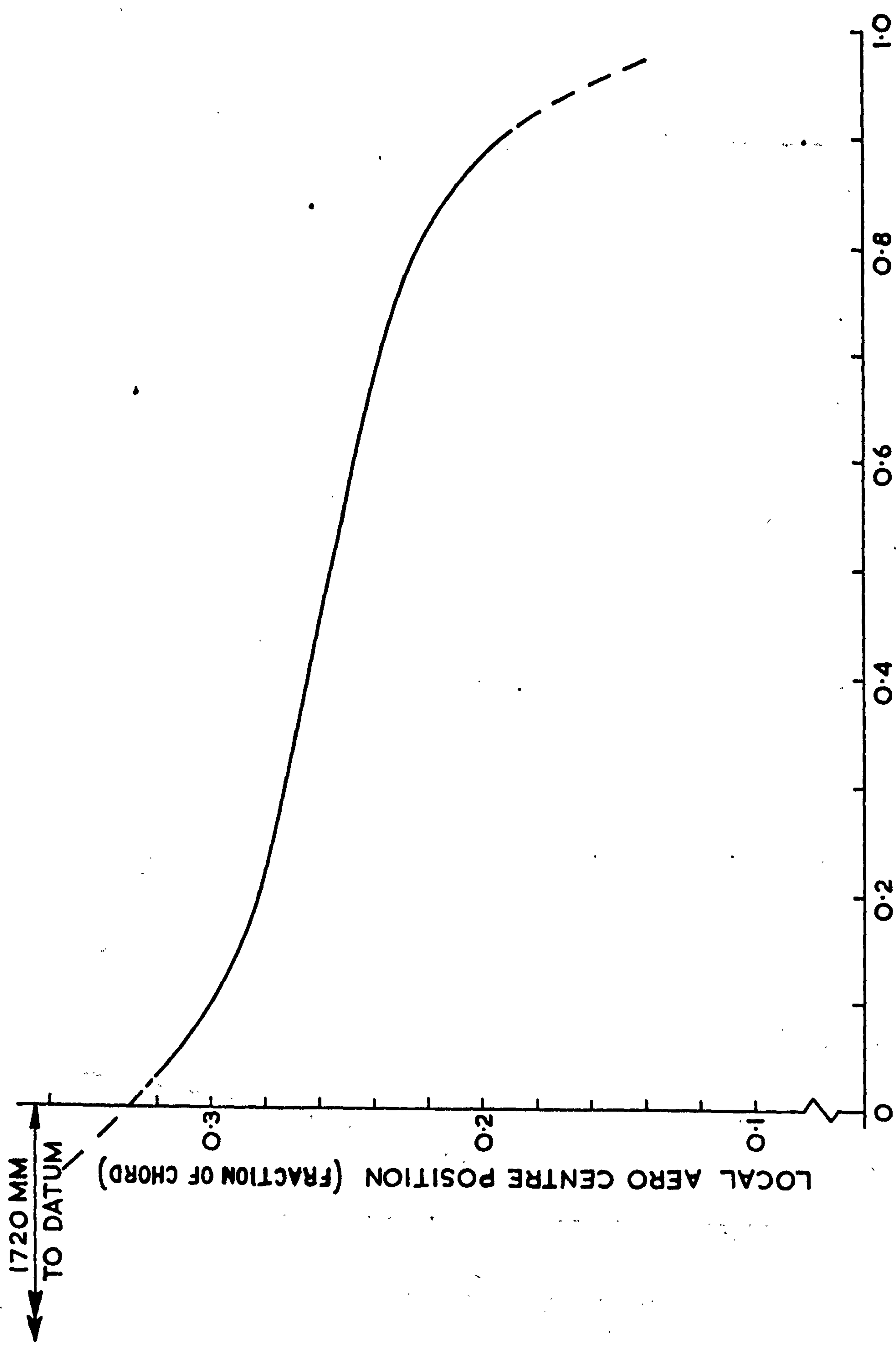


FIG. A74-13

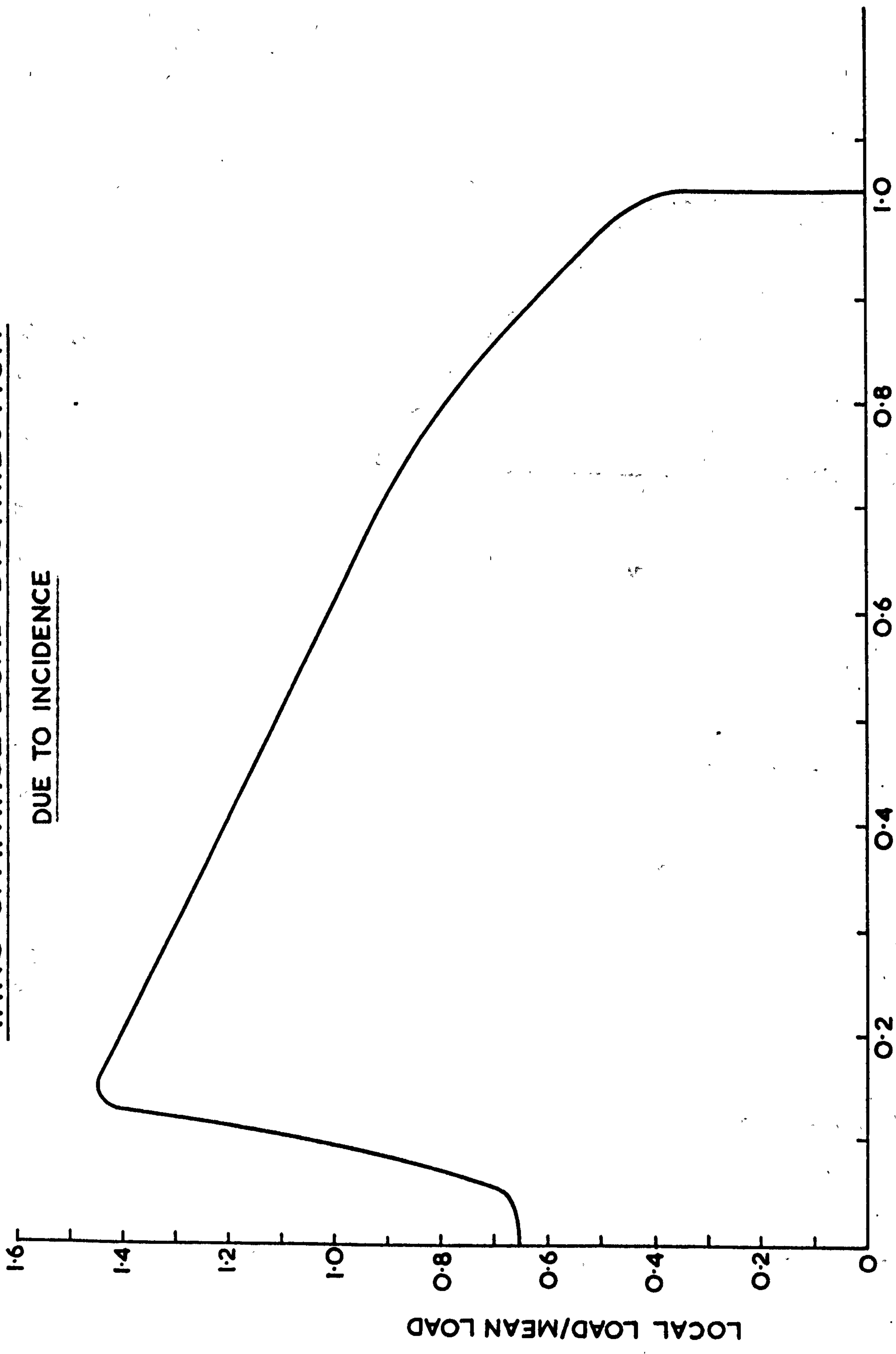
SPANWISE VARIATION OF FIN AERO CENTRE POSITION



HEIGHT (ABOVE 1720 MM CHORD)

WING SPANWISE LOAD DISTRIBUTION

DUE TO INCIDENCE



FRACTION OF SEMISPAN

WING SPANWISE LOAD DISTRIBUTION

DUE TO AILERONS, FLAPS, SLATS AND SPOILERS

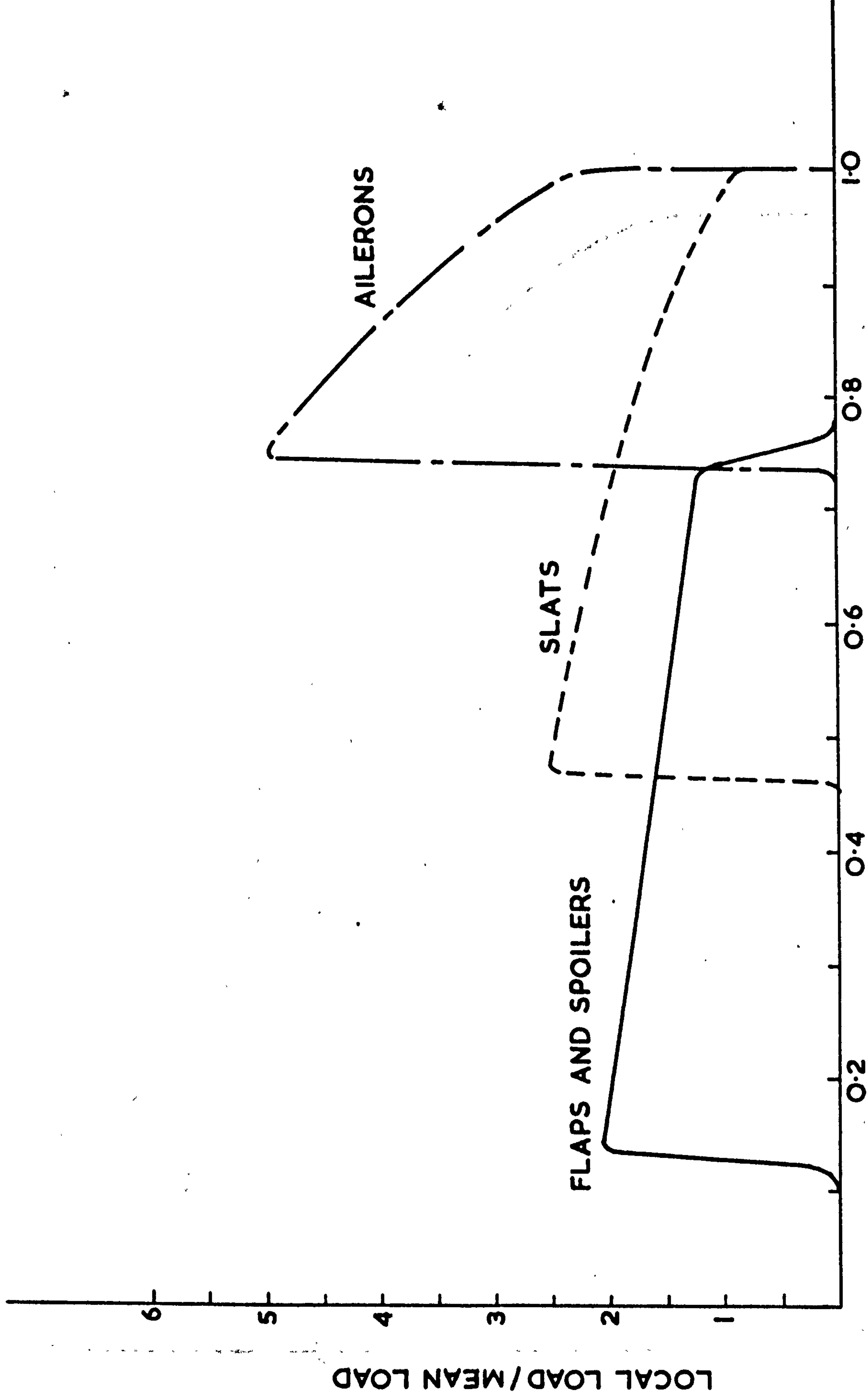
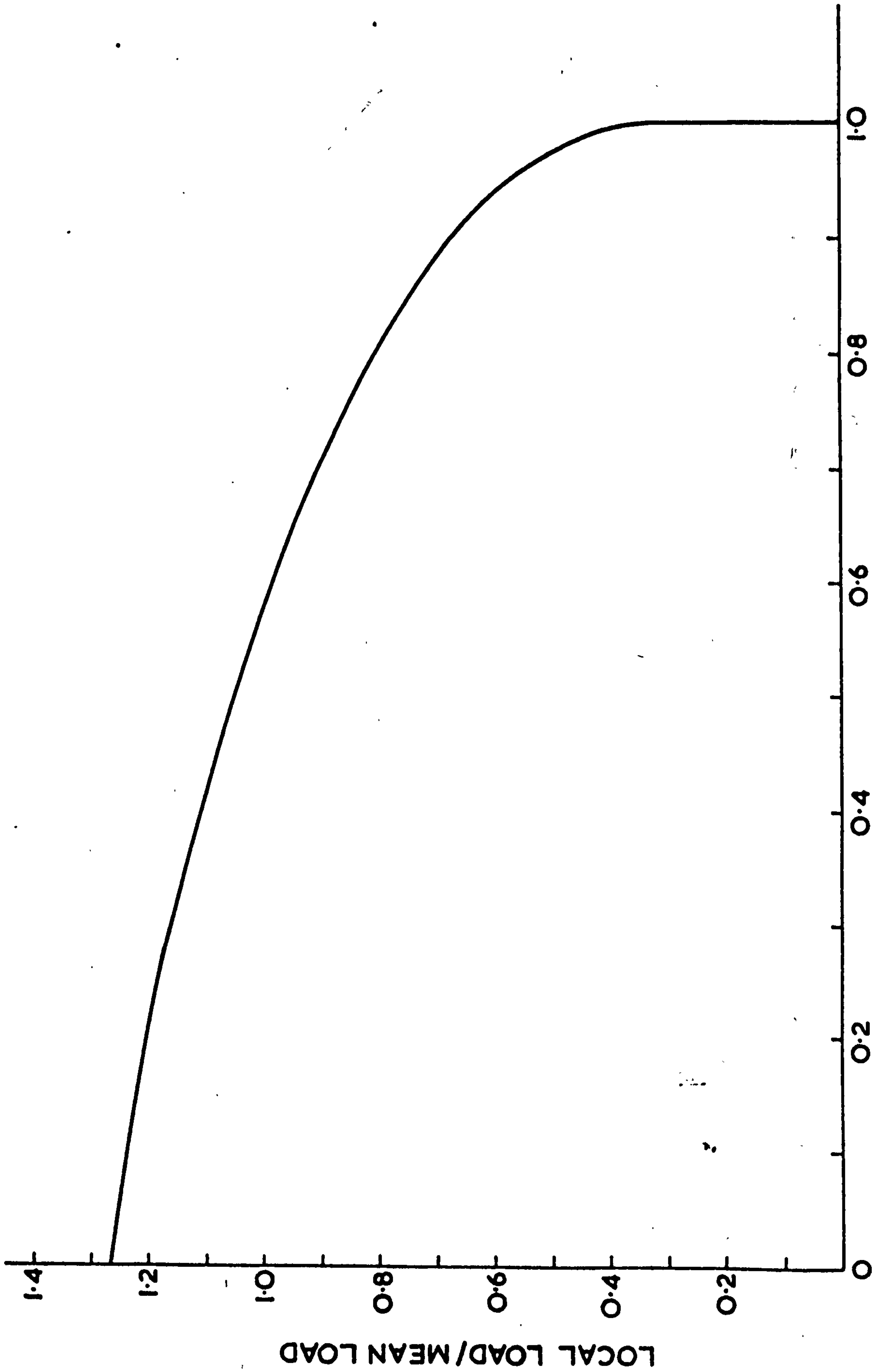


FIG A74-16

TAILPLANE SPANWISE LOAD DISTRIBUTION

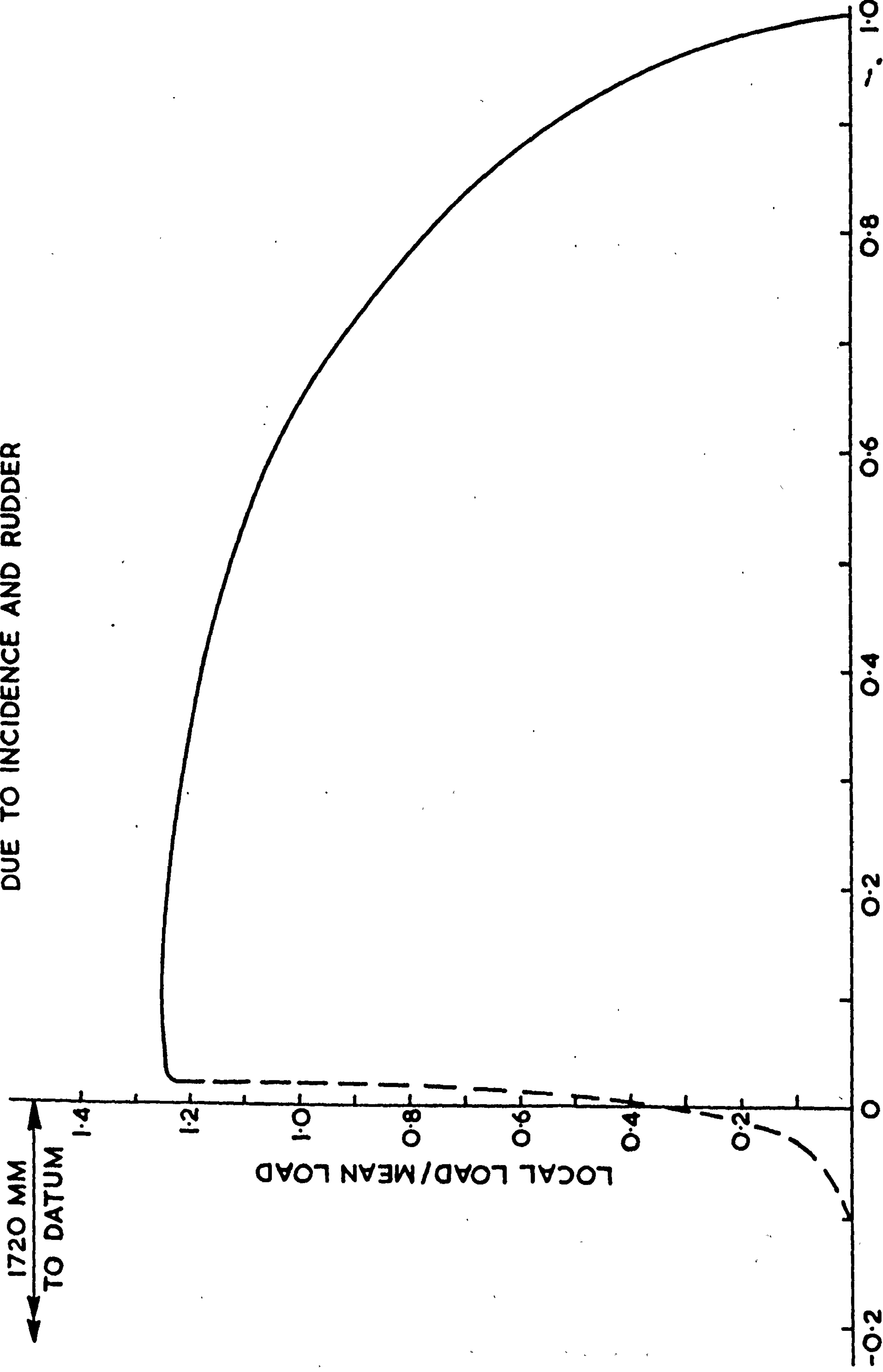
DUE TO INCIDENCE AND ELEVATOR



FRACTION OF SEMISPAN

FIN SPANWISE LOAD DISTRIBUTION

DUE TO INCIDENCE AND RUDDER



HEIGHT (ABOVE 1720 MM CHORD)

WING AND TAIL CHORDWISE LOAD DISTRIBUTION

DUE TO INCIDENCE
(IDEALISED)

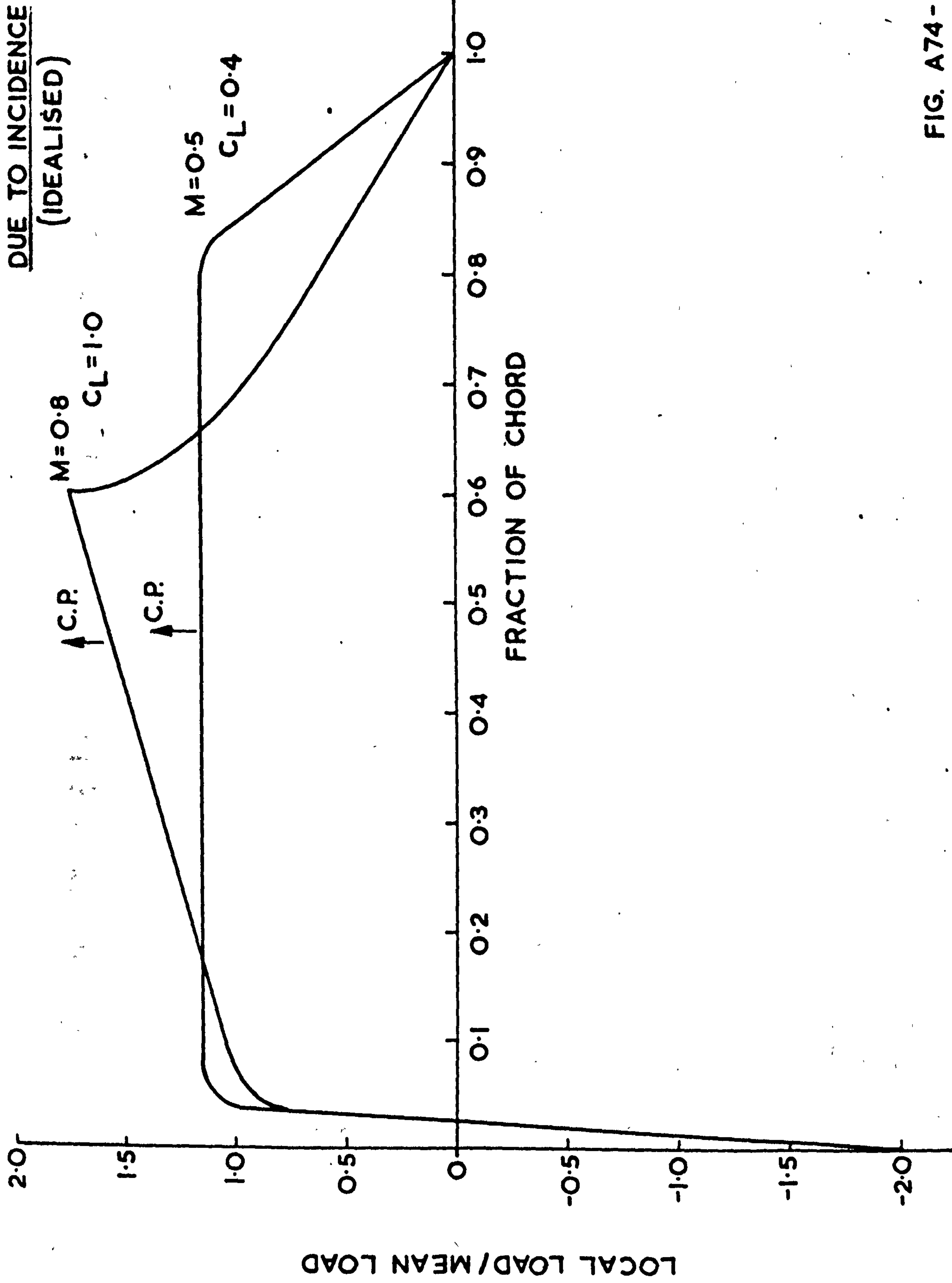


FIG. A74-19

WING CHORDWISE LOADING DUE TO FLAPS - TAKE OFF SETTING

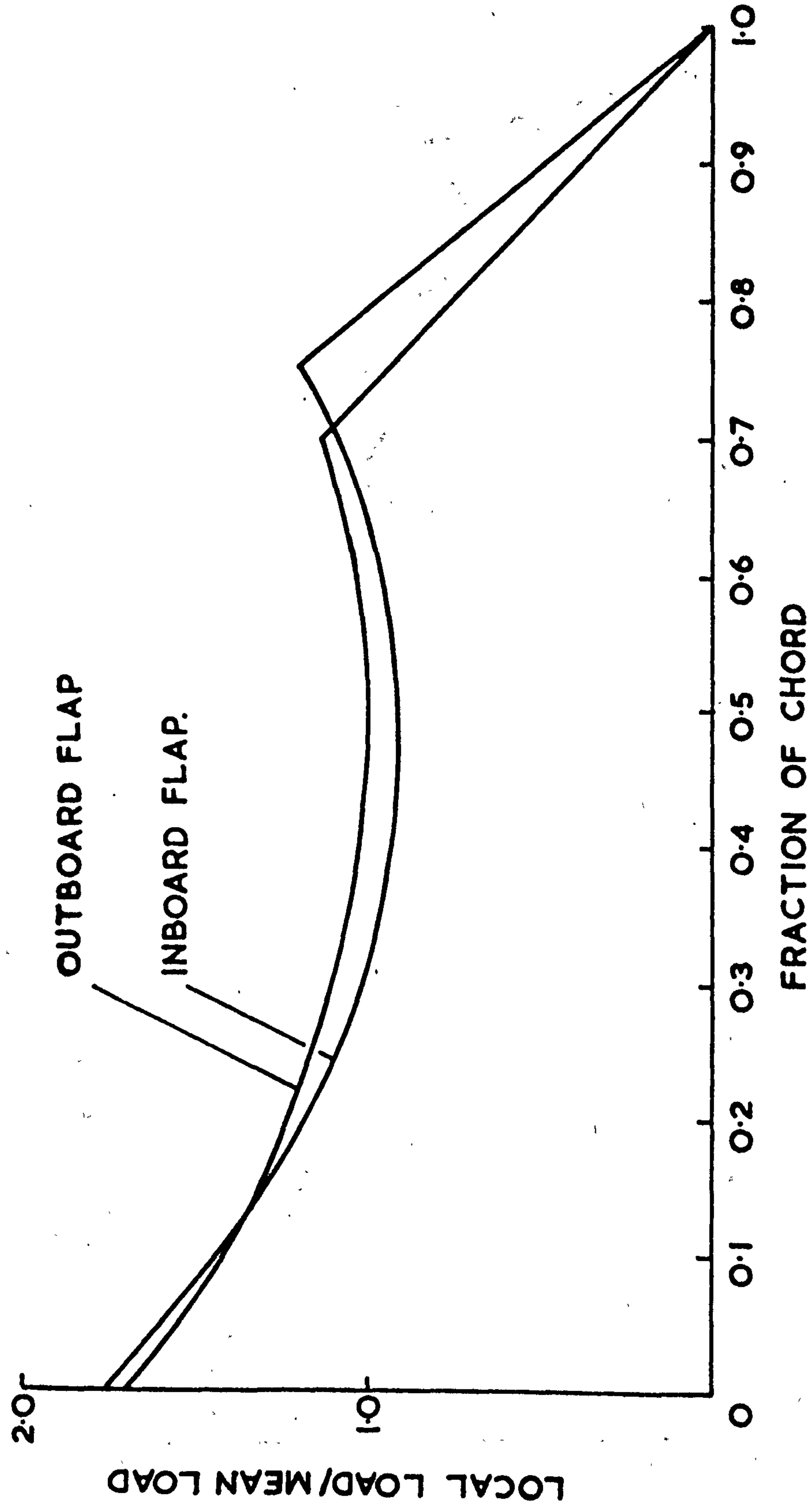


FIG A74-20

WING CHORDWISE LOADING DUE TO FLAPS - LANDING SETTING

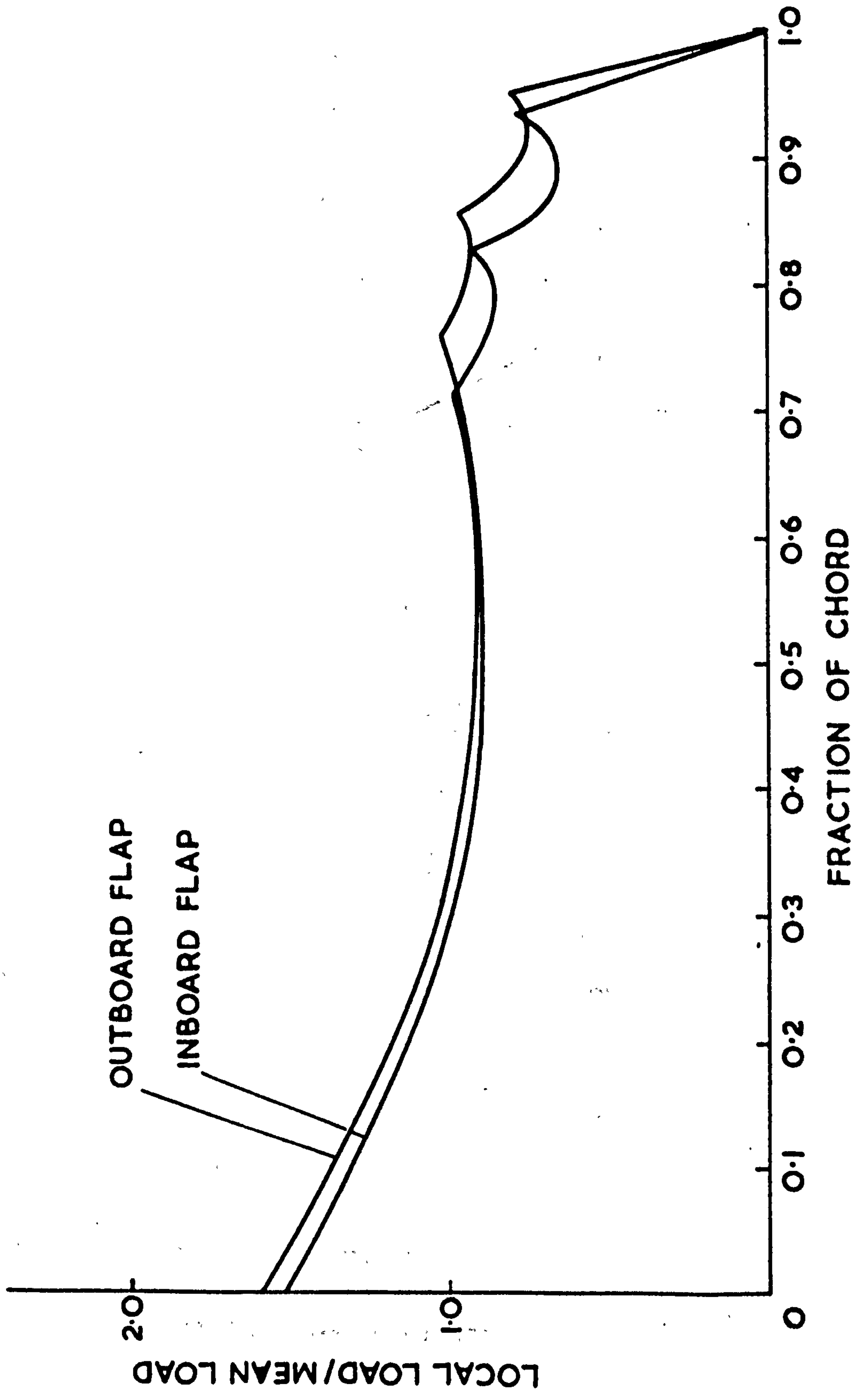


FIG A74-21

WING CHORDWISE LOAD DISTRIBUTIONS DUE TO SLATS AND SPOILERS

(BASED ON REFERENCE CHORD)

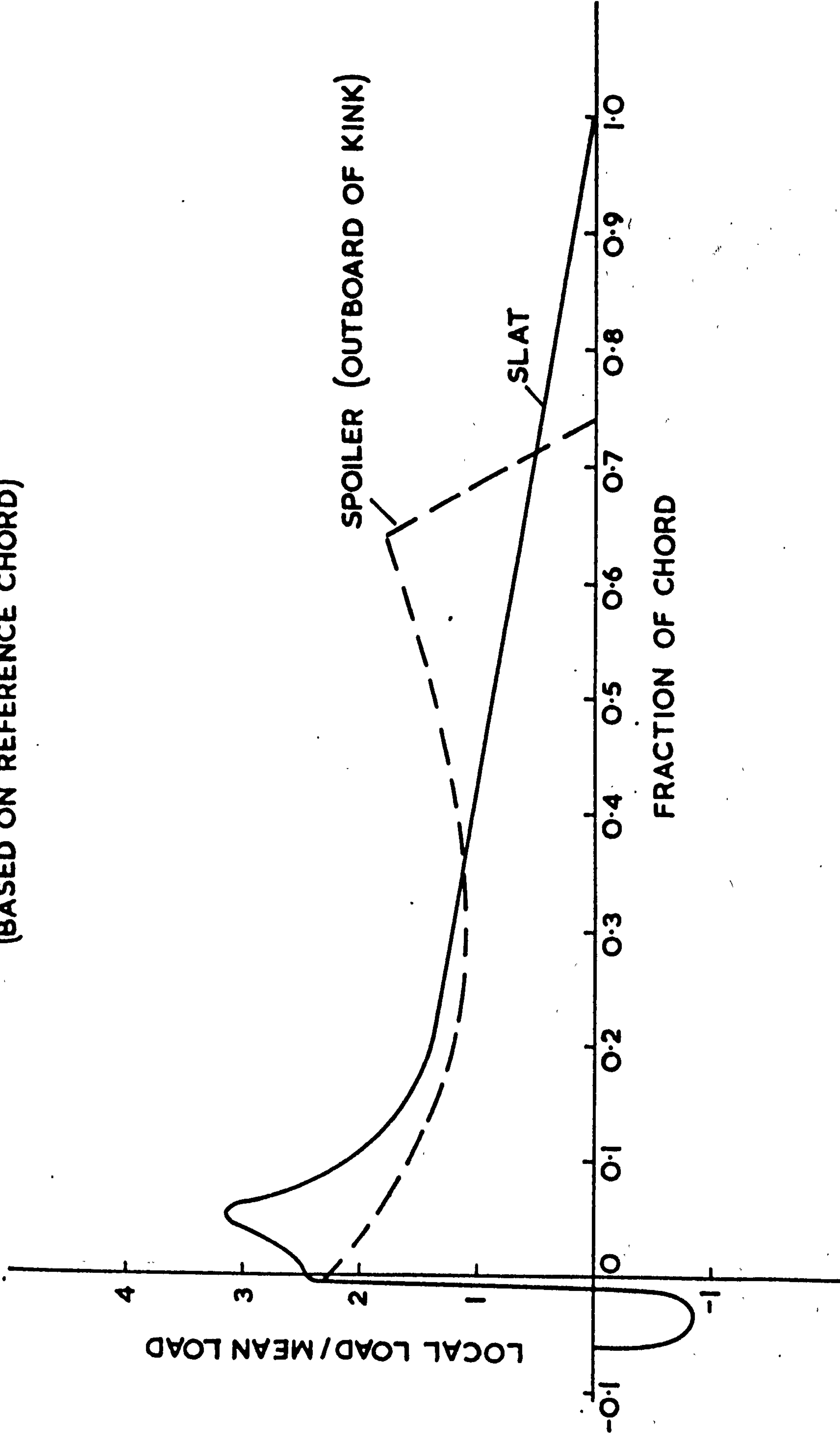


FIG A74-22

CHORDWISE LOAD DISTRIBUTIONS
DUE TO CONTROL DEFLECTIONS

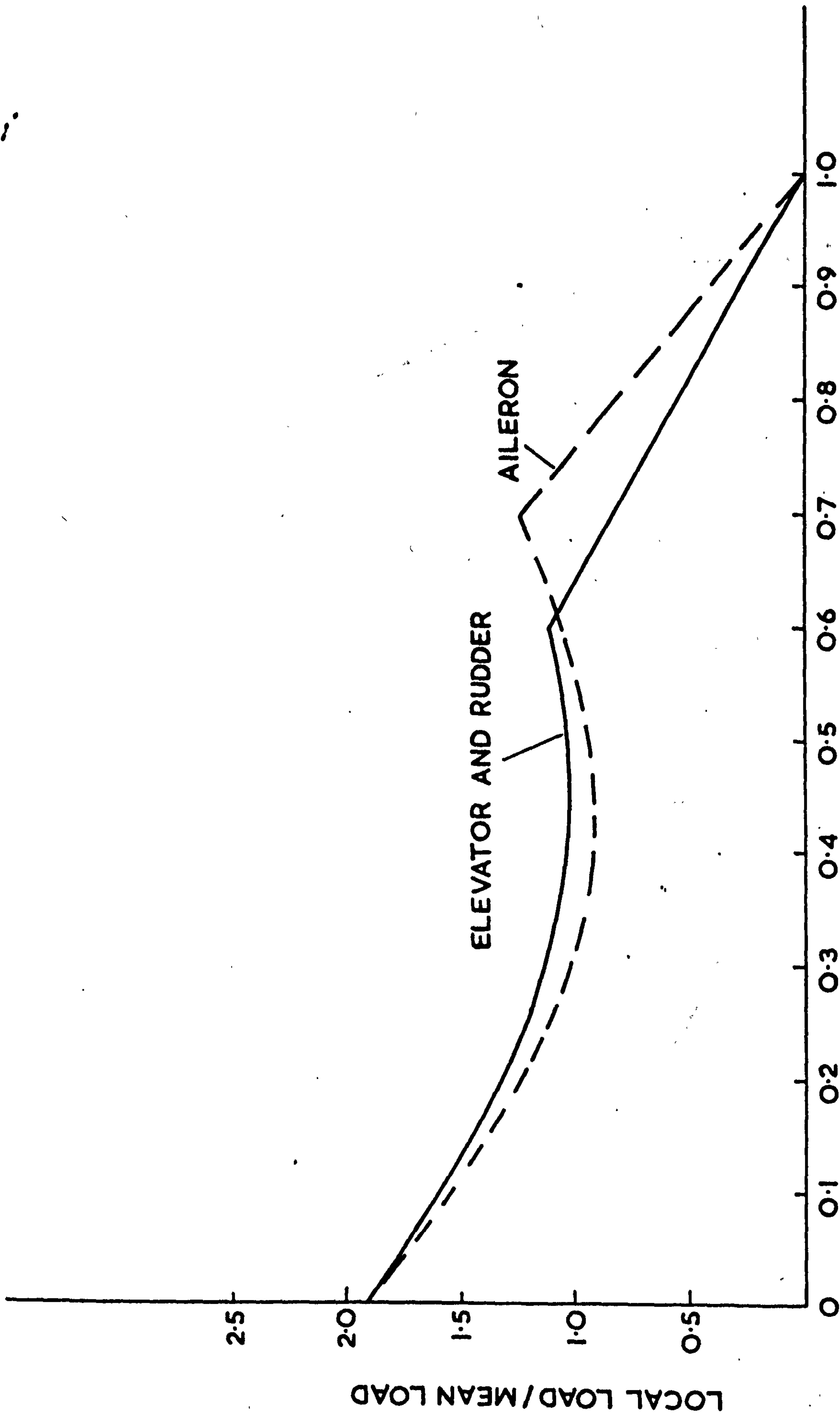
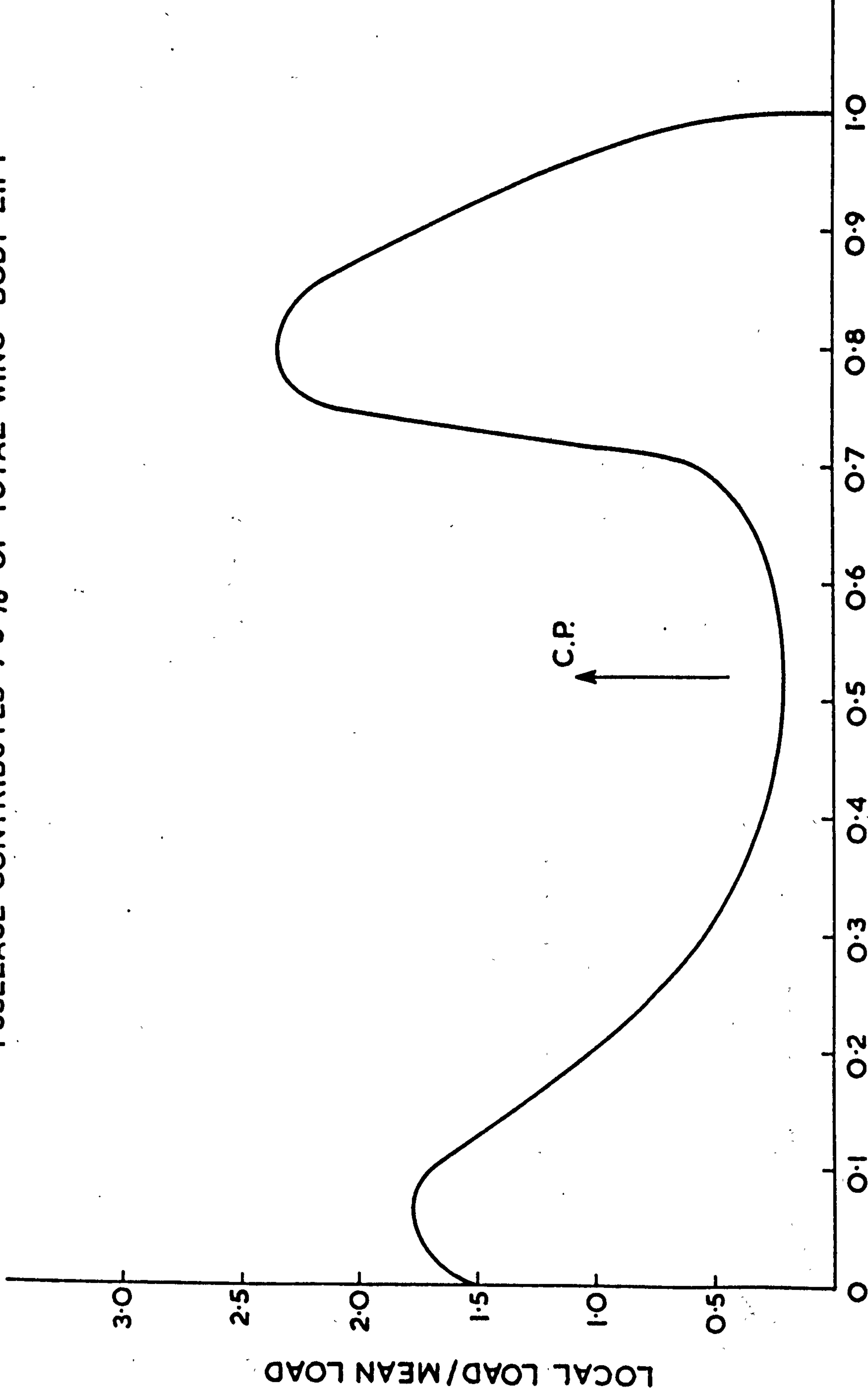


FIG. A74-23

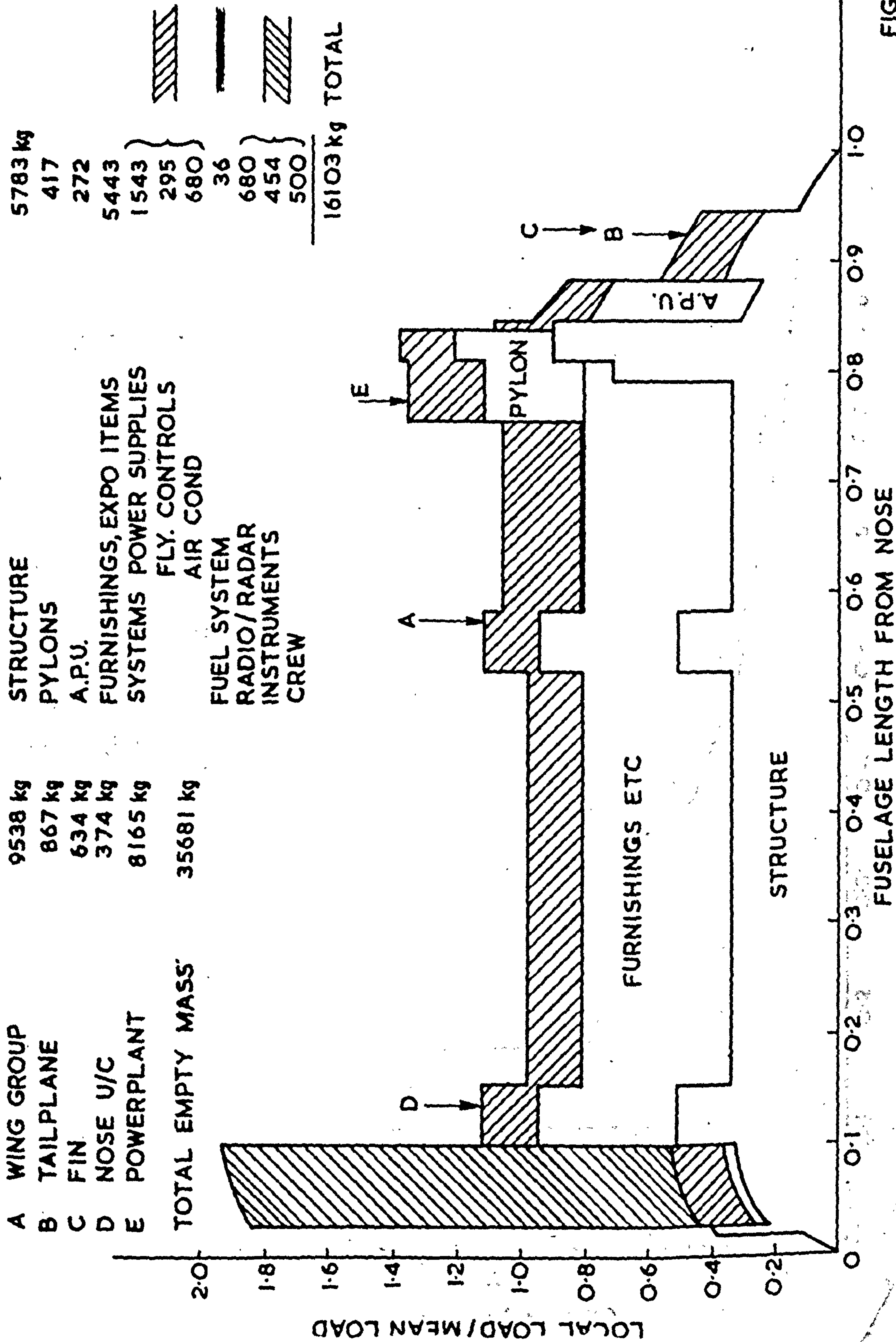
FUSELAGE LIFT DISTRIBUTION

FUSELAGE CONTRIBUTES 7.5% OF TOTAL WING-BODY LIFT



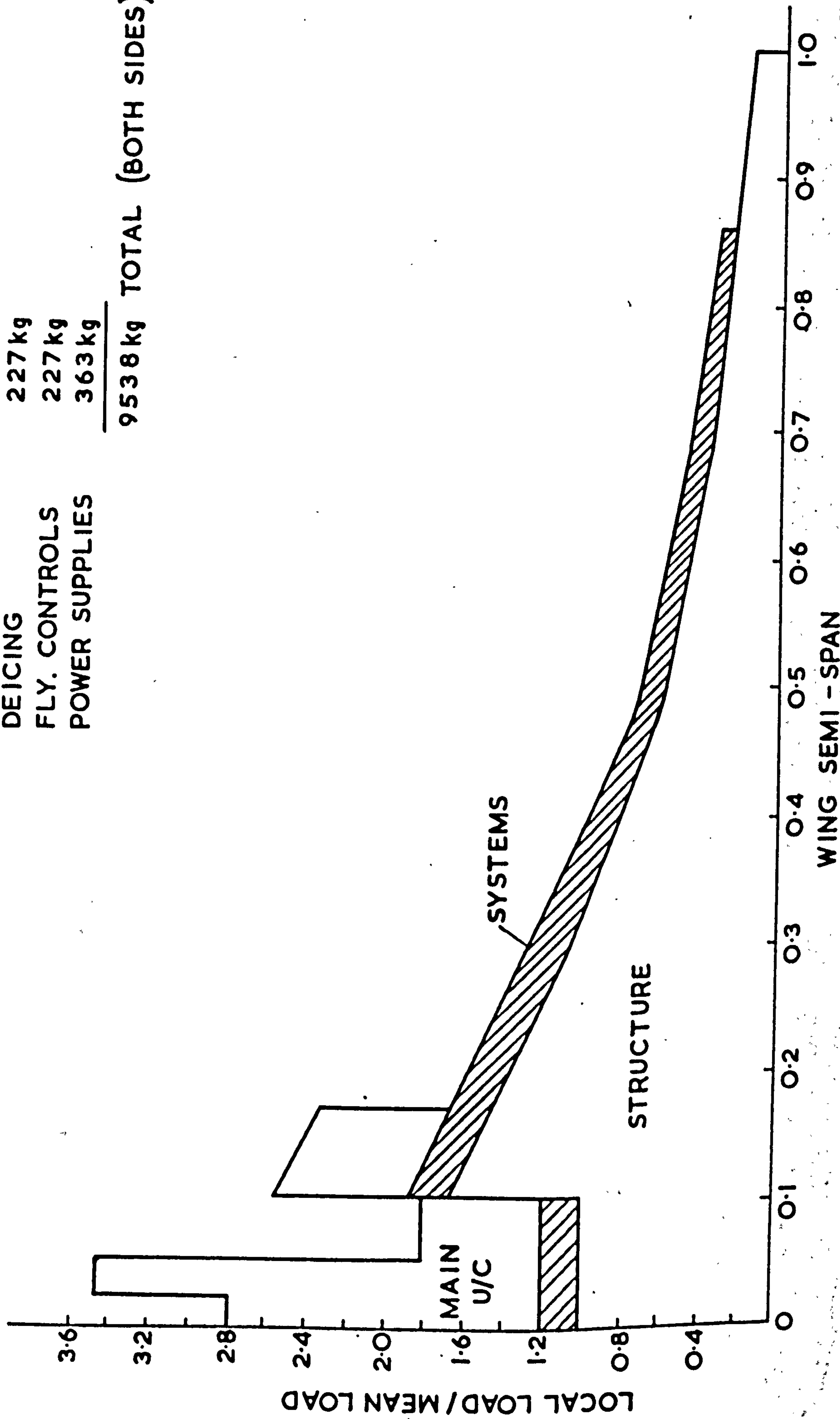
FRACTION OF FUSELAGE LENGTH

FUSELAGE INERTIA LOAD DISTRIBUTION



WING INERTIA LOAD DISTRIBUTION

STRUCTURE	6484 kg
MAIN U/C	1828 kg
SYSTEMS - FUEL	409 kg
DEICING	227 kg
FLY. CONTROLS	227 kg
POWER SUPPLIES	363 kg
TOTAL (BOTH SIDES)	9538 kg



FIN INERTIA LOAD DISTRIBUTION

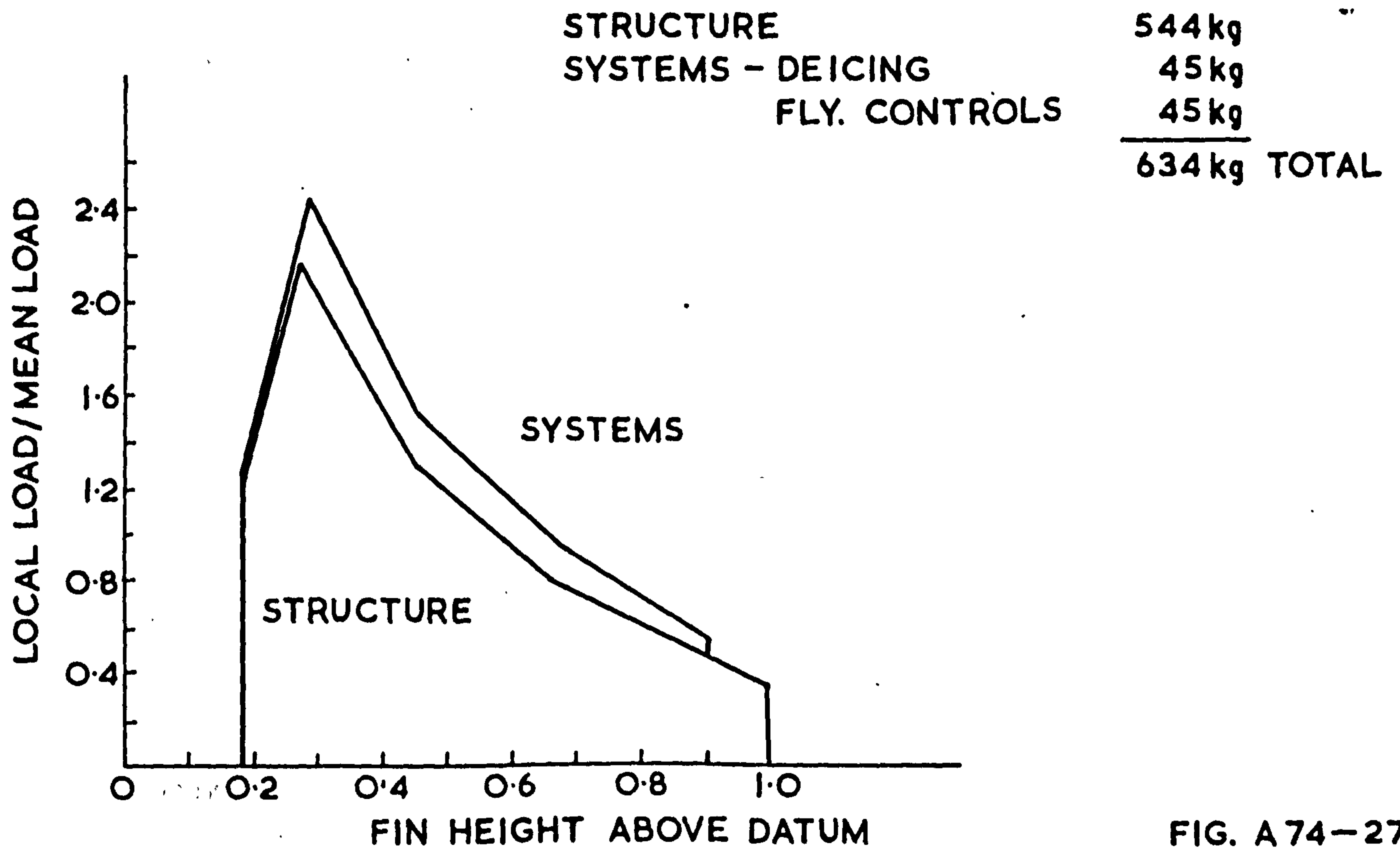


FIG. A74-27

TAILPLANE INERTIA LOAD DISTRIBUTION

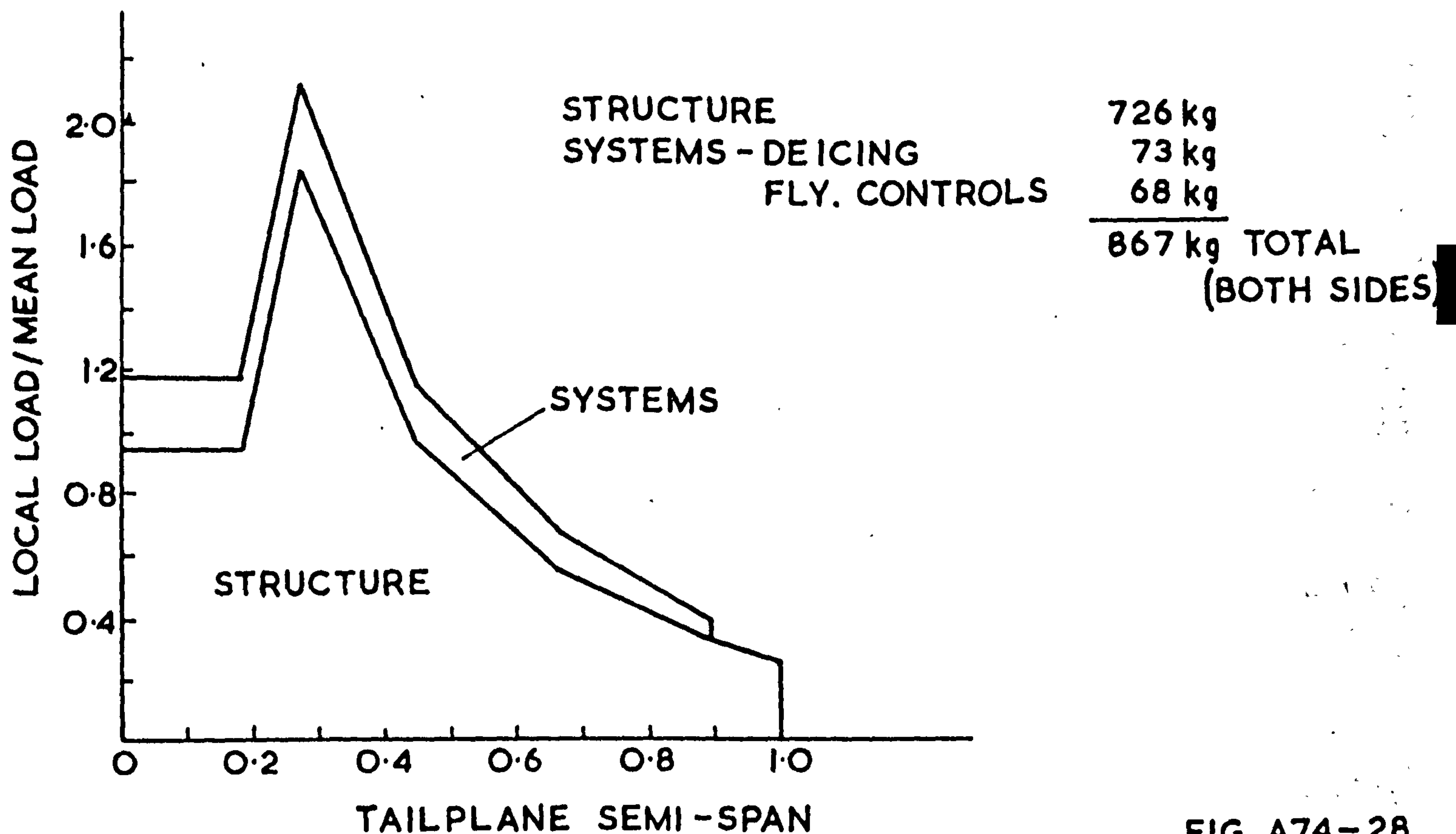


FIG. A74-28

PART 2

PUBLISHED REPORTS

(NOTE: As is usual in published papers these are presented in concise form. Whilst the philosophy, methods used, results and conclusions are given the extensive intermediate calculations required in each case have been omitted.)

Cranfield Report Aero No.18
April 1973

CRANFIELD INSTITUTE OF TECHNOLOGY

. COLLEGE OF AERONAUTICS

PERFORMANCE CHARACTERISTICS OF SHORT HAUL TRANSPORT
AIRCRAFT INTENDED TO OPERATE FROM REDUCED LENGTH RUNWAYS
by D. HOWE

SUMMARY

This report discusses the design characteristics of future short haul transport aircraft intended to operate from runways of reduced length relative to those used at the present time. Particular attention is paid to passenger comfort considerations and the influence these have on the take off and landing performance.

The results presented show that it should be possible to design reduced take off and landing (RTOL) aircraft to operate safely from runways of 4000 ft length without the need for power augmented lift. Such an aircraft would operate at speeds very similar to those used by current short haul transports, the main difference being in the need to provide a static thrust/weight ratio of the order of 0.4. On the other hand short take off and landing (STOL) aircraft intended to operate from runways of about 2000 ft length require a substantial degree of powered lift both for take off and landing. Installed thrust/weight ratio can be as high as 0.7 and the low approach speed of about 80 knots implies the possibility of serious low speed control difficulties.

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6. Effective mean design deceleration during braking.
7. Required approach lift coefficient - 250 knots EAS cruise
8. Required approach lift coefficient - 300 knots EAS cruise
9. Required approach lift coefficient - 350 knots EAS cruise.
10. Landing runway criterion - mechanical high lift devices.
11. Ground roll distance at take off.
12. Take off runway length criteria.
13. Mean braking deceleration during stopping after powerplant failure
- 14.) Take off distance as a function of cruise speed
- 15.) and unstick lift coefficient.
16. Take off distance as a function of thrust/weight ratio and unstick lift coefficient.
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19. Annual frequency of equivalent cross wind sideslip angles at Croydon.
20. Annual frequency of equivalent cross wind sideslip angles at Speke.

NOTATION

A	Aspect ratio
C_D	Drag coefficient
C_{LA}	Approach lift coefficient
C_{LM}	Part of lift coefficient produced by aerodynamic devices.
C_{LP}	Part of lift coefficient produced by powerplant
C_{LUS}	Unstick lift coefficient
D	Drag
L	Runway length
R	Bypass ratio of powerplant at cruise condition
T	Thrust
T_o	Static thrust
V	Velocity
V_A	Approach velocity
V_C	Cruise velocity
V_S	Stall velocity
V_{US}	Unstick velocity
V_1	Take off safety speed, all powerplants functioning
$(V_1)_F$	Take off safety speed with one powerplant failed
Z	Lift
a_1	Lift curve slope of wing
h	Vertical acceleration factor during initial climb out.
n	Number of powerplants
s_1	Ground roll distance corresponding to speed V_1 or $(V_1)_F$
s_2	Ground distance during climb to 35 ft, all powerplants functioning
\bar{s}_2	Ground distance during climb to 35 ft, one powerplant failed.
w	Take off wing loading
w_L	Minimum cruise/landing wing loading
$\Lambda_{1/2}$	Wing half chord sweep

1. INTRODUCTION

During the past two decades the improvements in the design of transport aircraft have been mainly in the direction of higher speed and greater payload. The convenience of air transport and the competitive economics resulting from design improvements have generated a rapidly expanding market. Passengers have come to expect a high standard of comfort, partly as a legacy of the first class image of early air transport and partly because of the smooth flight of high altitude turbine aircraft.

It would appear that air transport is entering a new phase of development. Until now the performance improvement and market expansion have been achieved without serious regard to the environment. Aircraft have been developed at the expense of ever increasing runway length and engine power. This has resulted in expanding noise footprints around aerodromes, a feature made worse by the increased frequency of flights as airport facilities develop towards maximum capacity. The situation is being reached where existing airports can no longer cope with traffic demands. The area sterilised or made unpleasant by an international airport is so large that new sites can only be located at inconveniently remote distances from the centres of population they are intended to serve. General congestion of surface transport aggravates this.

At the same time there is insistent public demand for a reduction of the nuisance of aircraft operating from existing aerodromes, particularly with respect to noise levels. In a recent lecture (Ref.1) Steiner suggested that the prime design considerations for future generations of transport would be noise, comfort and economics, in that order. This is a reversal of past practice where economic considerations have been dominant and have resulted in part in implied comfort improvements. Whilst noise levels have been of concern for some while improvement was not allowed to introduce a significant performance penalty. Safety considerations should perhaps be added to the list given by Steiner, since the increased frequency of operations and larger capacity aircraft must be accompanied by reduction in the accident rate if public confidence in air transport is to be maintained.

New generations of transport aircraft will have to be designed to take full account of environmental considerations. At the same time it will become increasingly more important to find ways of safely and efficiently handling the greater numbers of aircraft serving large centres of population. Proliferation of large airports is not feasible even where it is possible. Increase of capacity of existing facilities is limited by air traffic control and, possibly more significantly, by vortex wake decay from preceding aircraft. One of the ways of alleviating the congestion problem is the development of aircraft able to operate from airfields of very limited size located away from existing airports.

Short haul operations make up a large proportion of all flights and are frequently undertaken by smaller aircraft than those used for long range work. The ground delay associated with remotely located airports is also more significant in the case of shorter total journey time. Thus relatively small short range aircraft are the most likely candidates for operations from restricted airfields, particularly if these airfields can be located reasonably close to departure/destination areas. The transfer of the short haul operations away from the major airports would leave them relatively free for expansion of longer range operations by larger aircraft, and possibly in some cases short haul operations by very large aircraft where passenger density justifies this. A further bonus which would accrue from the development of this new class of short haul aircraft is that they could be used to serve the more remote points where airfield facilities are very limited. At the present time this sort of route employs small, slow aircraft which offer less than the generally acceptable comfort standards.

Current short haul transports with capacity for about 100 passengers require runway lengths of from 4500 ft to 6500 ft. The minimum flying speeds are such that cross wind effects are rarely of importance. Although some airfields with runways of this order of length do exist near to centres of population in both Europe and the United States most have already been developed into major airports. In many cases it is virtually impossible to find suitable new locations, especially bearing in mind the implied take off and landing flight paths and air traffic control restrictions. Therefore the proposed new class of short haul aircraft should be able to operate from smaller airfields.

The purpose of this report is to consider the primary requirements for this class of short haul aircraft and the interrelation of the main design parameters and characteristics.

1.1 Noise

The effect of noise level restrictions on the design of the aircraft is not easily evaluated. In the first place there is no really specific definition of the noise levels which will be regarded as acceptable. The United States FAR Part 36 requirements apply only to conventional aircraft, operated conventionally, and will probably prove to be no more than an interim solution. There have already been legal decisions in the United States in which restrictions have been imposed upon aircraft which meet Part 36. The minimum noise level limitations laid down by Part 36 apply to aircraft of 75,000 lb gross weight or less. The effective noise level below the flight path at 3.5 n.miles from take off must not exceed 93 PNdB for this weight although up to 108 PBdB is allowed for weights in excess of 600,000 lbs. During the approach to landing the comparable figures one n.mile from the threshold are 102 PNdB and 108 PNdB, with similar levels for a worst sideline condition. Some aircraft, such as the McDonnell Douglas DC10, Lockheed Tristar and Cessna Citation already achieve noise levels which are well below these limits.

Part 36 is perhaps best regarded as a device to ensure that the trend towards increasing noise levels was reversed in new designs. There is little doubt that each new type of aircraft will be expected to have a lower level of noise intrusion than its predecessors until such a time as aircraft noise is sensibly within the level of the surrounding ground environment. For urban locations this level will probably become about 80 to 85 PNdB and hence it will represent a severe design condition for aircraft. In practice in the first instance it is likely that the aim will be to keep the 90 PNdB ground footprint virtually within the confines of the airport, at least for smaller aircraft. In the United States the powerplant research aim is to reduce the noise level of aircraft relative to the latest ones in current service by about 10 PNdB. This would result in conventional aircraft of less than 200,000 lb gross weight more or less meeting the 90 PNdB target.

However even when it becomes possible to specify overall noise performance requirements it is still necessary to interpret these for design purposes. Whilst the powerplants are obviously the basic source of the aircraft noise there are many secondary but significant considerations. These include the method of operation of the aircraft, powerplant installation features and the nature of lift augmentation when this is employed. Some of these factors can only be treated subjectively until such a time as the design is actually tested. Others, such as the take off and approach flight conditions, can be analysed specifically. In general it can be said that the steeper the descent and take off paths and the shorter the runway requirement the smaller will be the noise footprint for a given aircraft noise characteristic. Thus these features are desirable from the noise standpoint, as is the reduction in installed thrust to the minimum consistent with performance requirements including climbout angle.

1.2 Passenger Comfort

One aspect of passenger comfort is associated with the furnishings, sound proofing and environmental control of the cabin. Much of this is in the hands of the operator and may be regarded as secondary in the context of the overall layout of the aircraft provided adequate weight and volume provision is made.

The other aspect of comfort is dependent upon the performance characteristics of the aircraft. Included in this category are the take off acceleration and apparent climb out angle, turbulence sensitivity in cruise, apparent descent angle, flare normal acceleration and braking deceleration. Passengers have come to expect certain standards in these respects and it is reasonable to presume that they would be unwilling to tolerate anything more severe than experienced with the best of present designs. Indeed the aim should be to improve upon these.

a) Take off and climb out

Forward facing seats are preferable for take off and climb out. The initial acceleration reassuringly forces the passenger back into his seat. Providing the rate of increase of acceleration is acceptable up to 0.5g or even possibly 0.6g initial acceleration may be tolerable. This is of considerable importance as the installed thrust/weight ratios necessary for operations from runways of reduced length are likely to be high. During climb out the effect of a steep apparent angle is merely to bring the passenger into a more reclined position, and quite large angles are acceptable. The limitation appears to be in respect of floor angle and is psychological.

b) Cruise

A simple way of interpreting the sensitivity of the aircraft to turbulence in cruise is in terms of the usual sharp edge gust relationship. This suggests that the comfort level is dependent upon the parameter:-

$$\left(\frac{a_1 V_c}{w_L} \right)$$

where V_c is the equivalent airspeed in cruise

w_L is the cruise wing loading

a_1 is the wing lift curve slope, and is primarily dependent upon wing planform shape for a given aerofoil section. Assuming a typical two dimensional aerofoil section property the value of a_1 is approximately given by

$$A / \left[0.32 + \frac{0.16A}{\cos \Lambda_{1/2}} \right]$$

where A is the aspect ratio

$\Lambda_{1/2}$ is the sweep of the mid chord line.

An analysis of a number of existing civil transport aircraft of all kinds suggests that for acceptable cruise comfort performance

$$\frac{a_1 V_c}{w_L} \leq 21$$

(V_c in knots EAS, w_L in lb/ft²)

Hence

$$w_L \geq \frac{A V_c}{21 \left(0.32 + \frac{0.16A}{\cos \Lambda_{1/2}} \right)} \quad \dots (1)$$

This relationship can be used to prescribe minimum values of cruise wing loading for a given planform and cruise condition. As a landing may occur more or less immediately after the end of the cruise it is reasonable to assume that the minimum cruise wing loading is equivalent to the maximum landing value. Actual values of w_L are shown in Figure 1, together with take off values based on the premise that for a short range transport the take off weight is approximately 1.15 times the maximum landing weight.

c) Approach and landing

The approach angle will not be greatly influenced by considerations of passenger comfort providing the actual attitude of the aircraft is not significantly removed from the horizontal. Aerodynamically this should be possible to arrange although a very steep descent could cause difficulties.

Of considerable importance are the normal acceleration during flare out, and the impact and horizontal decelerations as the aircraft is brought to rest. Normal accelerations of 0.25g are only rarely exceeded in transport operations, with 0.15g being a much more typical value. It is reasonable to assume that a maximum value of up to 0.25g may be tolerated during flare out at the end of the approach path or during the actual landing impact.

The longitudinal deceleration during the ground run is of particular importance as the landing distance required by the aircraft is critically dependent upon it. In practice it varies substantially during stopping due to the interaction of various effects such as aerodynamic drag, reverse thrust and ground friction coefficient. It is convenient to discuss the problem in terms of a mean value providing it is understood that the peak value does not exceed this by too great a margin. The assumption of conventional forward facing seats for the passengers has already been assumed in order to tolerate the expected take off performance. Neither passengers nor the operators are likely to be prepared to accept the inconvenience and weight penalties which arise if full restraint harness is fitted, in spite of the fact that a lap-diagonal arrangement is now readily accepted in automobiles. Some evidence of the deceleration which passengers are prepared to expect in normal circumstances can be obtained from investigations undertaken for the Japanese National Railways (Ref.2) and by General Motors (Ref.3). In the former case the sample population consisted of university students and in the latter technicians who might reasonably be expected to be more tolerant to acceleration levels than, say, the elderly. The results are summarised in Figure 2, and in interpreting them for airline operations certain points must be borne in mind.

Airline passengers will expect to notice some deceleration during landing and may well become alarmed if they do not. They will be sitting in well designed, comfortable seats with some degree of restraint. Taking the mean of the population and the 'slightly uncomfortable' rating of 4 as an indication of what might be tolerated suggests that it would be unwise to assume a mean stopping deceleration greater than about 0.22g.

An examination of the stated landing performance of current airliners (Ref.4) indicates that the majority of them use mean design deceleration values of between 0.3g and 0.37g. In deriving these values it would appear that the only consideration is the deceleration which is physically obtainable under the design conditions. However it must be remembered that the quoted runway lengths for landing are in fact obtained by factoring the ideal distance by 1.67 to allow for operational variations and contingencies. In a normal landing this margin is unnecessary and the pilot has more stopping distance available than that used to deduce the mean decelerations. It can be shown that in practice the maximum stopping distance available to the pilot can be up to almost twice the design value and hence theoretically the mean deceleration could be reduced to as low as 0.16g to 0.17g in many landings. Of course even in a perfect landing the pilot will not use the whole of the available stopping distance and if an average distance usage of 0.75 of the maximum available is assumed the mean stopping deceleration is found to be in the range of 0.21g to 0.23g. This suggests that current aircraft actually use longitudinal decelerations which are of the same order as the tolerable comfort level, and therefore that it would be unwise to exceed the implied design value of about 0.33g in new designs. On the other hand it is possible that improved methods of bringing the aircraft to rest could be used to give higher available deceleration levels during adverse conditions such as wet runways and an argument might then possibly be made for some reduction of the 1.67 runway safety factor.

1.3 Economics

Economic considerations must always rate highly in the assessment of the operators. Of the various items which contribute to direct operating cost the first and fuel costs are of particular interest. First costs have risen steeply in recent years as a result of greater design sophistication and general inflationary trends. It is now becoming increasingly difficult to finance the purchase of new aircraft. Whilst noise and passenger comfort may be short term overriding considerations it will be necessary to find ways to stabilise first cost.

Fuel costs have steadily become less important during the development of jet transport aircraft. There are two reasons for this. Firstly the price of fuel has fallen due to the greater volume used and secondly engine specific fuel consumption has been reduced by some 35 per cent. This latter

trend is partly the result of increased engine bypass ratio and hence further developments in this respect to reduce noise will give additional benefit, possibly resulting in a further 15-20 per cent reduction in specific fuel consumption. Unfortunately the trend towards reduction of fuel price has now been reversed and the signs of a potential fuel shortage are beginning to be apparent. There are a number of reasons why sharp increases of fuel price can be expected in the future. Demand is increasing rapidly and with some of the producer countries now importing oil to meet their own requirements a sellers market exists. Further the likely requirements a decade hence are so great as to seriously threaten to exhaust in a short time the known worldwide reserves of easily obtainable fuel, and conservation of resources is most likely to be achieved by cost control. Therefore it may be concluded that fuel costs will become of ever increasing importance to the economics of the aircraft and could become the dominant design criterion once noise targets have been achieved.

2. PERFORMANCE CONSIDERATIONS

2.1 Approach speed and descent angle

It is important that the approach speed of an airliner should be as high as possible within the limitations imposed by available runway length and air traffic control. There are three main reasons for this.

- a) The higher the approach speed the less significant are the effects of crosswind which is a particularly important consideration when operation from single runway airfields is considered.
- b) The control power of conventional aerodynamic devices falls rapidly with reduction of approach speed.
- c) The wing loading should be as high as possible for reasons of cruise comfort, range performance and weight and this implies a high approach speed for a given lift coefficient.

For a given runway length three major parameters determine the maximum tolerable approach speed.

- 1) Glide path angle. Conventional aircraft approach along a 3° glide slope and increase of this angle enables some decrease of runway length to be obtained. Further a steeper descent path reduces the size of the noise footprint. Evidence suggests that the real limit on glide angle is established by the vertical rate at which pilots are prepared to descend during the final stage of landing. A figure of 1000 ft/min has been suggested when a conventional flare out procedure is used. The resulting relationship between speed and tolerable descent angle is shown in Figure 3.

This clearly shows that unless the approach speed is low, or the restriction can be removed the allowable increase in descent angle is small. The vertical rate of descent limitation could be overcome by a fully automatic landing or perhaps by designing the aircraft to withstand the impact of landing without the need to flare. With high angles of descent there could be a problem of achieving a correct balance of forces on the aircraft and maintaining the cabin floor more or less horizontal. This is illustrated by Figure 4 but is possibly more serious in the early stages of descent. There may well also be air traffic control restrictions if mixed traffic is operated from the same airfield. The different angles of descent used would have to be sufficiently separated from one another to enable the aircraft to be brought in from different circuits or stacks. The required difference in glide slope angle is likely to be at least 4° which is equivalent to a horizontal spacing of about 4 miles at 2000 ft altitude.

- ii) Flare and touchdown. The flare conventionally initiated at the end of the approach is intended to reduce the actual vertical impact velocity to zero but during experiments with steep descents some pilots have expressed a preference for eliminating it (Ref.5). This avoids the problem of judging when to start the flare and enables the pilot to aim more accurately at a specific point on the runway. It has a secondary advantage in further slightly reducing the landing distance and may also enable higher rates of descent to be tolerated. The limitations are established by the compromise between the acceptable undercarriage shock absorber stroke and passenger response to impact deceleration.
- iii) Stopping deceleration. This has already been discussed above in paragraph 1.2(c). If a value of 0.17g mean deceleration over the total factored stopping distance is assumed together with, for example, a 0.25g flare and impact at 4 ft/sec vertical velocity the maximum approach velocity as a function of runway length and descent angle is as given in Figure 5. The 4 ft/sec impact velocity has been assumed somewhat arbitrarily. The 1000 ft/min rate of descent limitation is indicated and the corresponding mean deceleration over the unfactored stopping distance is shown in Figure 6.

2.2 Landing wing loading and approach lift coefficient

By assuming the approach speeds given in Figure 5 and the wing loadings of Figure 1 which are based on the cruise comfort criterion it is possible to evaluate the approach lift coefficient. Because of the definition of the cruise comfort criterion, Eq.1, for a given approach speed the wing loading and hence lift coefficient is a function of cruise speed and wing planform. The approach lift coefficient

required to enable an aircraft to land on a runway of given length is shown in Figures 7 to 9 for specific cruise speeds of 250, 300 and 350 knots EAS respectively. These clearly show that in order to keep the wing loading and braking deceleration down to the levels suggested by comfort requirements it is necessary to use high values of approach lift coefficient for STOL type operations on runways of the order of 2000 ft length. If it is assumed that conventional mechanical high lift devices can be used to enable approach lift coefficients in the range of 1.8 to 2.0 to be achieved with the usual 1.3 speed margin relative to the stall then the appropriate runway lengths are shown in Figure 10. This suggests that aircraft using mechanical high lift systems could be developed to enable them to land on runways of from 3000 ft to 4000 ft in length. For example when a lift coefficient of 1.8, corresponding to a maximum usable value of about 3, is associated with a cruise speed of 300 knots EAS and an equivalent unswept aspect ratio of 8 a runway length of about 3500 ft is required if the approach path angle is $7\frac{1}{2}^\circ$ or 3800 ft if it is 3° . Application of the 1000 ft/min rate of descent limitation suggests a maximum tolerable descent angle of 5° in this case but this is not especially critical in terms of runway performance. The corresponding approach speed would be approximately 109 knots with assumed landing and corresponding take off wing loadings of about 72 and 83 lb/sq ft respectively.

3. PERFORMANCE CONSIDERATIONS - TAKE OFF

3.1 Take off requirements

The runway length required for the take off may be determined by any of three considerations:-

- a) The distance required to lift off and climb to 35 ft altitude when all the powerplants are functioning normally, factored by 1.15.
- b) The total distance determined by the necessity to bring the aircraft to rest after a powerplant failure at the decision speed. The decision speed is determined in association with c) below.
- c) The total distance necessary to climb to 35 ft altitude after a powerplant failure at, or above, the decision speed. This case is thus related to b) and taken together they determine the decision speed.

The main difficulty associated with determining the take off distance arises from the possible interaction of thrust and lift in those cases where power augmented lift is used. A completely general analysis is complex but a reasonable indication of the requirements can be ascertained if certain simplifying assumptions are made. In particular it will be assumed that there is zero longitudinal acceleration immediately after lift off so that the initial climb out speed is equal to the take off safety speed.

In general the available lift coefficient will decrease in the event of an engine failure. Let the lift coefficient at the take off safety speed with all powerplants functioning, V_1 , be $(C_{LM} + C_{LP})$ where the subscripts M and P refer to the contributions from the aerodynamic and powered contributions respectively. After a failure of one out of a total of n powerplants the lift coefficient is approximately reduced to:-

$$\frac{(n-1)}{n} C_{LP} + C_{LM}$$

To give the same total lift as previously the speed must be increased to:-

$$(V_1)_F = V_1 \left[\frac{C_{LP} + C_{LM}}{\left(\frac{n-1}{n}\right) C_{LP} + C_M} \right]^{1/2} \dots (2)$$

where $(V_1)_F$ is the safe take off speed with one powerplant failed and will correspond to a ground roll of s_1 , say. The total take off runway distance is s_1 plus the ground distance covered during the climb to 35 ft altitude. When all the powerplants are functioning the vertical acceleration can be higher than in the failed case since the lift coefficient is (C_{LP}/n) greater. Let the distance from lift to the 35 ft altitude be s_2 . Then the runway length must be at least:-

$$L = 1.15(s_1 + s_2)$$

Alternatively if a powerplant fails at $(V_1)_F$, the aircraft must be able to stop or climb out to 35 ft altitude. To achieve the latter within the all powerplants functioning take off distance, L , the climb out distance must not exceed:-

$$\bar{s}_2 = 1.15s_2 + 0.15s_1 \dots (3)$$

When the stopping distance is the more critical then the situation is complicated by the need to consider a decision speed below $(V_1)_F$ and the effect of failure between this speed and $(V_1)_F$.

3.2 Unstick lift coefficient and ground roll

Since the forward speed at 35 ft altitude condition must be 1.2 times the stalling speed as it is the take off safety speed, then as a result of the assumption of zero longitudinal acceleration during the initial climb out phase

$$(V_1)_F = 1.2V_s$$

where V_s is the appropriate stalling speed, in this case with a powerplant failed.

However in those cases where there is a longitudinal acceleration during the initial climb out the unstick speed, V_{US} , must not be less than $1.05V_S$ and hence as a conservative value:-

$$V_{US} = 0.875(V_1)_F \quad \dots (4)$$

In assessing the ground roll to lift off, s_1 , it is necessary to make allowances for the rolling resistance, aerodynamic drag and variation of thrust with forward speed. In normal circumstances it is sufficient to assume that the first of these is equivalent to a friction coefficient of 0.03. The last two are rather more difficult to deal with. As far as the drag is concerned the magnitude of the coefficient is primarily a function of the status of the high lift devices. For simplicity it will be assumed here that the induced drag effects are small during the ground roll, that is the high lift devices are not deployed completely until just prior to rotation. In this case a somewhat arbitrary value of drag coefficient is assumed to give a value of 0.001 for the parameter (C_D/w) . The thrust variation with forward speed during take off is mainly determined by the powerplant bypass ratio. An examination of typical engine characteristics (Ref.4) has suggested that for speeds up to 120 knots:-

$$T = T_0 [1 - 8.5(1+0.13R)V \times 10^{-3}] \dots (5)$$

where T is the thrust corresponding to a velocity, V knots, T_0 being the static value and R the bypass ratio defined at the cruise condition.

Using these assumptions the ground roll distance can be estimated as a function of velocity, static thrust/weight ratio and powerplant bypass ratio. A typical set of results is shown in Figure 11, where s_1 is shown for the case of $R = 10$ with some values for $R = 5$ superimposed. These latter values show the secondary effect of bypass ratio at lower velocities.

3.3 Climb out

The value of unstick velocity, V_{US} , given by Eq.(4) implies an available normal acceleration of up to $0.3g$ at $(V_1)_F$, the lift off speed, as opposed to a margin of only $0.1g$ when $V_{US} = 1.05V_S$, even when a powerplant has failed since $(V_1)_F$ is based on this condition. The latter value of normal acceleration may be taken as an absolute minimum available. In practice allowance for time to deploy high lift devices is necessary and the normal acceleration assumed can make provision for this.

The time taken to reach 35 ft altitude is $(70/h)^{1/2}$ where h is the mean vertical acceleration, that is it is not likely to exceed about 4.7 seconds. The ground distance covered in this phase is therefore

$$\begin{aligned} \left(\frac{70}{h}\right)^{1/2} (V_1)_F &= s_2 \text{ ft} && \text{when all powerplants are functioning} \\ &= \bar{s}_2 \text{ ft} && \text{when a powerplant has failed.} \end{aligned} \quad (6)$$

3.4 Take off runway length

The lift off speed is directly related to the parameter (C_{LUS}/w) by Eq.(4). Figure 12 is a presentation of the factored take off distances required for a bypass ratio 10 powerplant in terms of this parameter. To evaluate the results the sum of $(s_1 + s_2)$ from Figure 10 and Eq.(6) has been factored by 1.15. Figure 13 shows the mean braking decelerations which have to be achieved to enable the aircraft to be brought to rest in the corresponding distance \bar{s}_2 as given by Eq.(3) after a powerplant failure at speed $(V_1)_F$. Thus cross reference between Figures 12 and 13 for any given acceptable value of mean braking deceleration enables the validity of the derived take off lengths to be checked. For example if it is assumed that in emergency conditions the tolerable mean braking deceleration is 0.5g, then only those results above the validity boundaries shown in Figure 12 are realistic. Below the validity lines the take off length is underestimated within the assumptions made. The additional distance required is readily deduced from Figure 13 by comparison of the achievable and required decelerations.

Since the take off wing loading has been derived in Figure 1 as a function of cruise speed and wing planform it is possible to deduce the required unstick lift coefficient. For example Figures 14 and 15 show the take off distances as a function of wing planform characteristics for a thrust/weight ratio of 0.5 and mean climb out vertical acceleration of 0.1g and 0.2g respectively. The effect of unstick lift coefficient and design cruise speed is shown. Figure 16 shows the effect of thrust/weight ratio for given values of C_{LUS} and the case of a design cruise speed of 300 knots EAS. In this instance the range of unstick lift coefficient of from 1.4 to 1.8 has been chosen to indicate the performance likely to be possible for an aircraft using only mechanical high lift devices, bearing in mind the desirability of keeping climb out lift/drag ratio as high as possible. Finally Figure 17 presents the required thrust/weight ratio for given values of C_{LUS} , a design cruise speed of 300 knots EAS, climb out vertical acceleration of 0.2g and specific runway lengths of 2000 ft and 4000 ft. These lengths have been selected to represent conditions appropriate to STOL and RTOL operations respectively. Figure 14 to 17 are all based on the use of a powerplant with a bypass ratio of 10 and no

emergency braking restriction has been applied.

A general summary of these results is that STOL operations from runways of 2000 ft length require unstick lift coefficients of the order of three associated with thrust/weight ratios in excess of 0.5. Mechanical high lift devices used alone should enable an RTOL aircraft to operate safely from a 4000 ft runway with an unstick lift coefficient of less than 2.0 and a thrust/weight ratio of the order of 0.4.

4. LOW SPEED CONTROL

Investigation of the low speed control characteristics of a particular STOL aircraft (Ref.6) suggests that the major low speed control problems are associated with powerplant failure and cross wind landing.

4.1 Powerplant failure

The importance of powerplant failure in influencing the low speed control requirements is dependent upon the layout and number of the engines and the degree of powered lift. Thus it is not possible to generalise other than to comment that as the flight speed is reduced the powerplant failure case becomes more significant and is likely to become critical, especially for double failure on approach.

4.2 Crosswind landing

The use of an R/STOL transport from single runway aerodromes introduces the possibility of a severe cross wind landing control problem. The aircraft must be able to approach in a specified mean cross wind and retain sufficient control power to be able to cope with gusting about that mean. Whilst various cross wind approach techniques may be used, a good idea of the severity of the problem can be gauged by the magnitude of the equivalent sideslip angle. The variation of the equivalent sideslip angle with cross wind and approach speed, is shown in Figure 18. The magnitudes of cross wind quoted have been chosen to coincide with the standard wind velocity groups in meteorological tables.

In order to obtain an indication of the significance of the single runway cross wind landing case the record of wind velocities at two locations have been analysed (Ref.7). Purely for comparative purposes the sites chosen were Croydon and Speke. The former is considered to be representative of a relatively sheltered inland site and the latter an exposed coastal site. If it is assumed that the single runway is orientated to minimise the effects of the most severe cross wind conditions the annual occurrences of cross winds, given as a function of equivalent sideslip angle are as shown for the two sites in Figure 19 and 20.

Two boundaries are shown in these diagrams. The 12.5° angle may be regarded as reasonably typical of current practice. On the other hand the 20° boundary is representative of what may be achieved with a specially designed aircraft. The special provisions could well include main undercarriage steering. As can be seen there is little restriction on operations carried out under conditions similar to those experienced at Croydon. Even with the 12.5° limitation and an approach speed as low as 90 knots there are only about 100 hours a year on average when approaches would be precluded. When the conditions experienced at Speke are more typical it is necessary to increase the approach speed to about 120 knots before obtaining the same conditions. It is apparent that a 20° equivalent sideslip condition is only likely to be necessary when the approach speed is about 80 knots or less. A 20° angle is likely to give rise to an adverse passenger reaction.

The investigation undertaken by Ward (Ref.6) suggests that the lateral gust velocity is just less than half of the mean wind speed. In the case of Speke with the most favourable runway orientation the maximum lateral gust velocity was found to be about 13 knots. This roughly corresponds to the equivalent mean wind condition sideslip angle of 15° at 100 knots approach speed.

5. DISCUSSION

The results presented show clearly the importance of passenger comfort in determining certain vital design parameters. For example it is not possible to use take off wing loadings of much less than 70 lb/sq ft even for moderate cruise speeds and a value of over 80 lb/sq ft is likely to be more typical in practice. Similarly the design braking deceleration on landing is unlikely to exceed 0.35g.

As far as steep descent during approach is concerned the main problem is associated with the rate at which the ground is approached. There is little to be gained here except for low approach speed conditions unless the use of automatic flight control or some other device can be used to remove the 1000 ft/min rate of descent restriction at present imposed by pilot opinion. Any steep descent requires a high effective drag to be developed with the aircraft in a more or less horizontal attitude for passenger comfort. This could cause difficulties or at least require the use of special devices.

The analysis of take off performance in a general way is complex but some indication of likely criteria is presented in Figures 11 to 16. Summarising these it can be stated that take off from relatively short runways necessitates high static thrust/weight ratio. For example this parameter can be expected to be in excess of 0.4 for safe operation from a 4000 ft runway and as much as 0.7 for 2000 ft runway operations. The former of these two is associated with an unstick lift coefficient which could be achieved by using mechanical high lift devices, but the latter certainly requires the use of some powered lift.

Landing behaviour is more readily dealt with and in general is less critical than take off in terms of runway length. As can be seen by reference to Figures 5 to 10 the employment of only mechanical high lift devices enables use to be made of runways of about 3500 ft length with a corresponding approach speed in excess of 100 knots. An approach speed of 80 knots or less is necessary for 2000 ft runway operation and this is associated with approach lift coefficients of 3.0 or more.

Comparison of the take off and landing figures shows that mechanical high lift devices should enable transport aircraft to be operated from runways of 4000 ft or more, with take off being critical. The thrust/weight ratio would be about 0.4 and the approach speed almost 120 knots. If an RTOL aircraft is defined as one which does not primarily use powered lift to improve low speed performance then 4000 ft can be regarded as the approximate lower bound of RTOL performance. For operation from shorter runways some degree of powered lift is necessary during take off although landings without powered lift are possible on runways of 3500 ft length or even somewhat less. This brings the aircraft into the STOL regime although true STOL performance is probably nearer to 2000 ft runway operation. For this case a considerable degree of powered lift is required with both unstick and approach lift coefficients of the order of three or more. Static thrust/weight is likely to be well in excess of 0.5 and the approach speed about 80 knots. Flight speeds as low as this introduce significant low speed control problems, especially those associated with cross wind landing onto single runway airports. In these circumstances it is relevant to question the case for an STOL design and to consider whether a VTOL or near VTOL concept where virtually all the lift is derived from the powerplants is not more logical. Cross wind landing and high rate of descent difficulties together with passenger comfort restrictions on wing loading and acceleration are all virtually removed. Against this it must be stated that the built in thrust needs to be about twice as great with the consequent effect on noise level, fuel consumption and first cost.

6. CONCLUSIONS

Several important conclusions can be drawn from the results derived in this study.

1. Passenger comfort considerations impose overriding limitations which have a significant effect upon the performance of transport aircraft designed to operate from runways of reduced length.
2. In general take off rather than landing performance determines the length of runway required for safe operation.
3. It should be possible to design RTOL aircraft which do not require powered lift augmentation, for operation from runways of 4000 ft length.

4. For operation from runways of less than 4000 ft some degree of lift augmentation is necessary, although if landing is the only criterion this is not required for runways in excess of about 3500 ft.
5. True STOL operation from runways of around 2000 ft length necessitates a substantial amount of powered lift. The thrust/weight ratio required is well in excess of 0.5 and approach speeds are of the order of 80 knots. This in turn introduces significant low speed control difficulties.
6. Noise limitations will introduce severe requirements in the design of future transport aircraft and it will be essential to arrange the layout and operation to alleviate these as much as possible.

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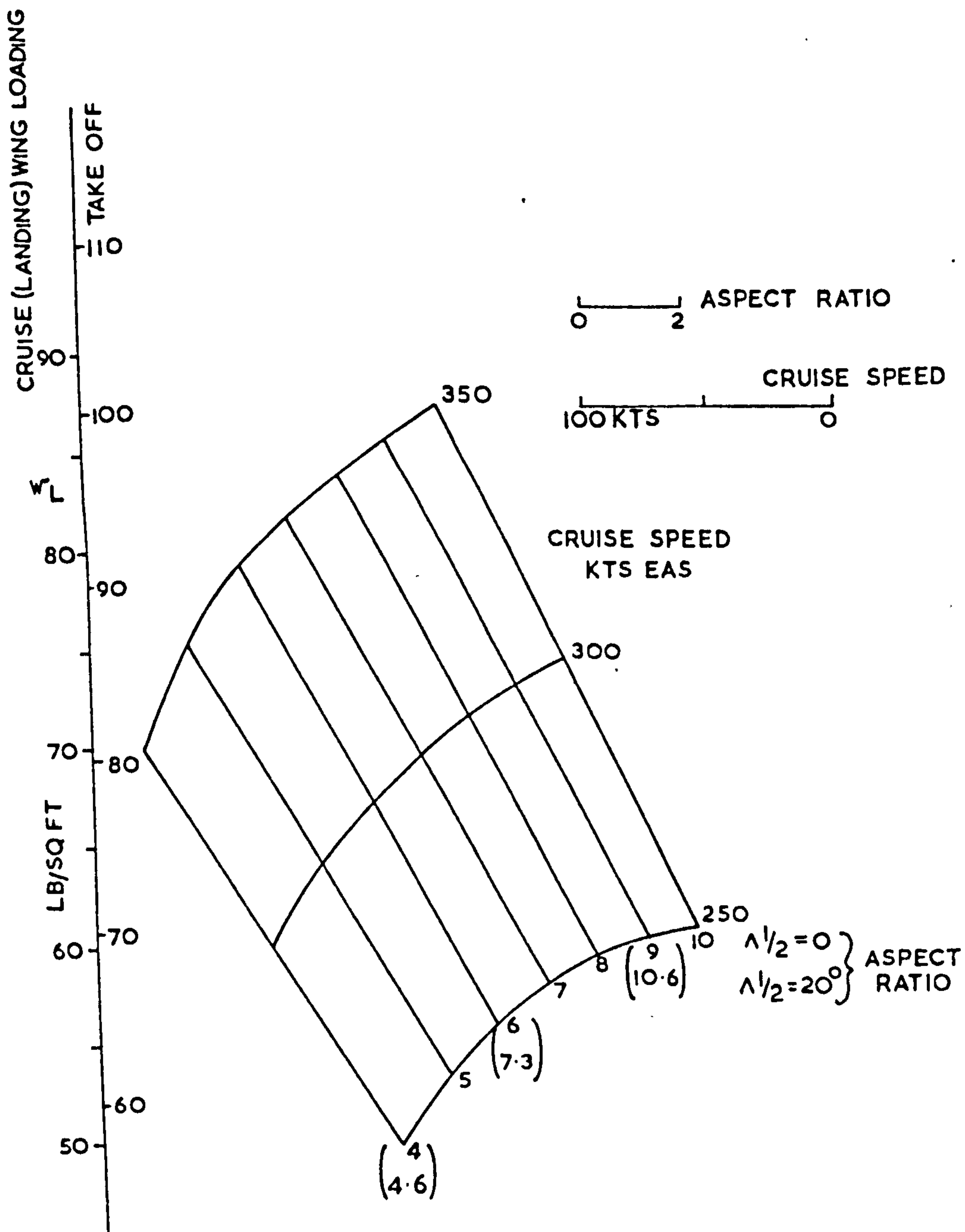


FIG. 1. MINIMUM WING LOADING FOR CRUISE COMFORT

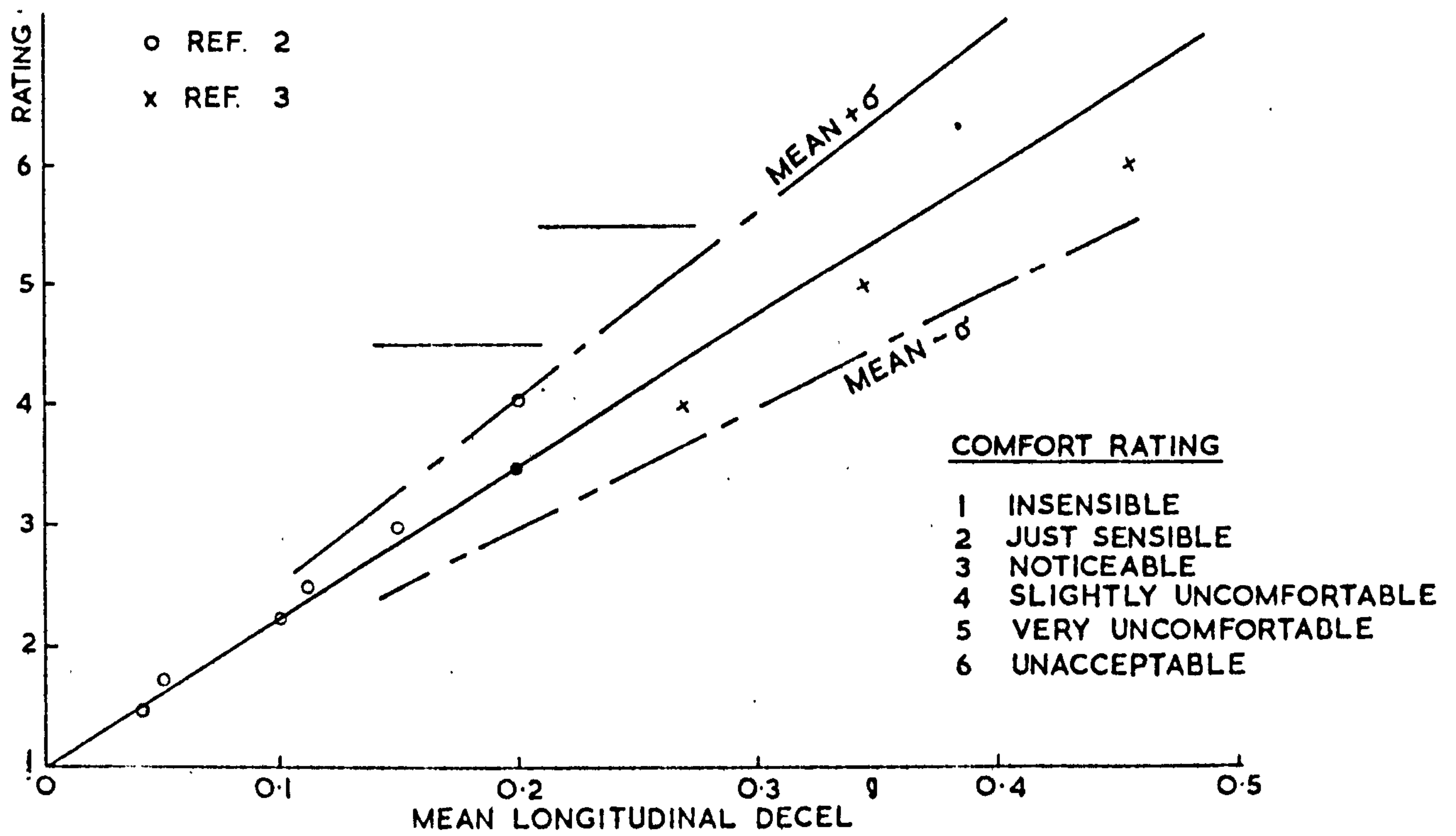


FIG. 2. COMFORT RATING FOR MEAN LONGITUDINAL DECELERATION

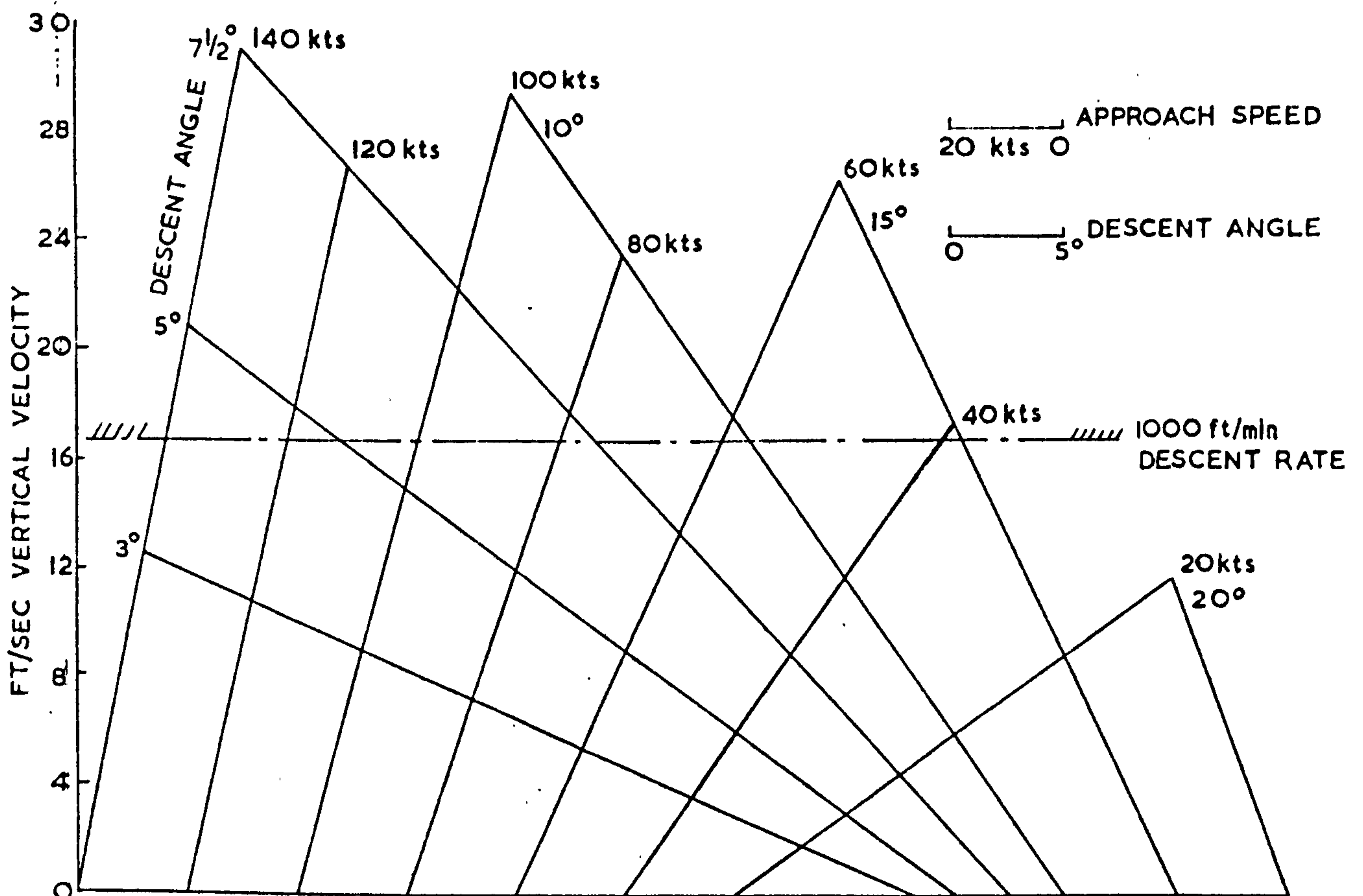


FIG. 3. VARIATION OF VERTICAL DESCENT VELOCITY WITH APPROACH ANGLE AND SPEED

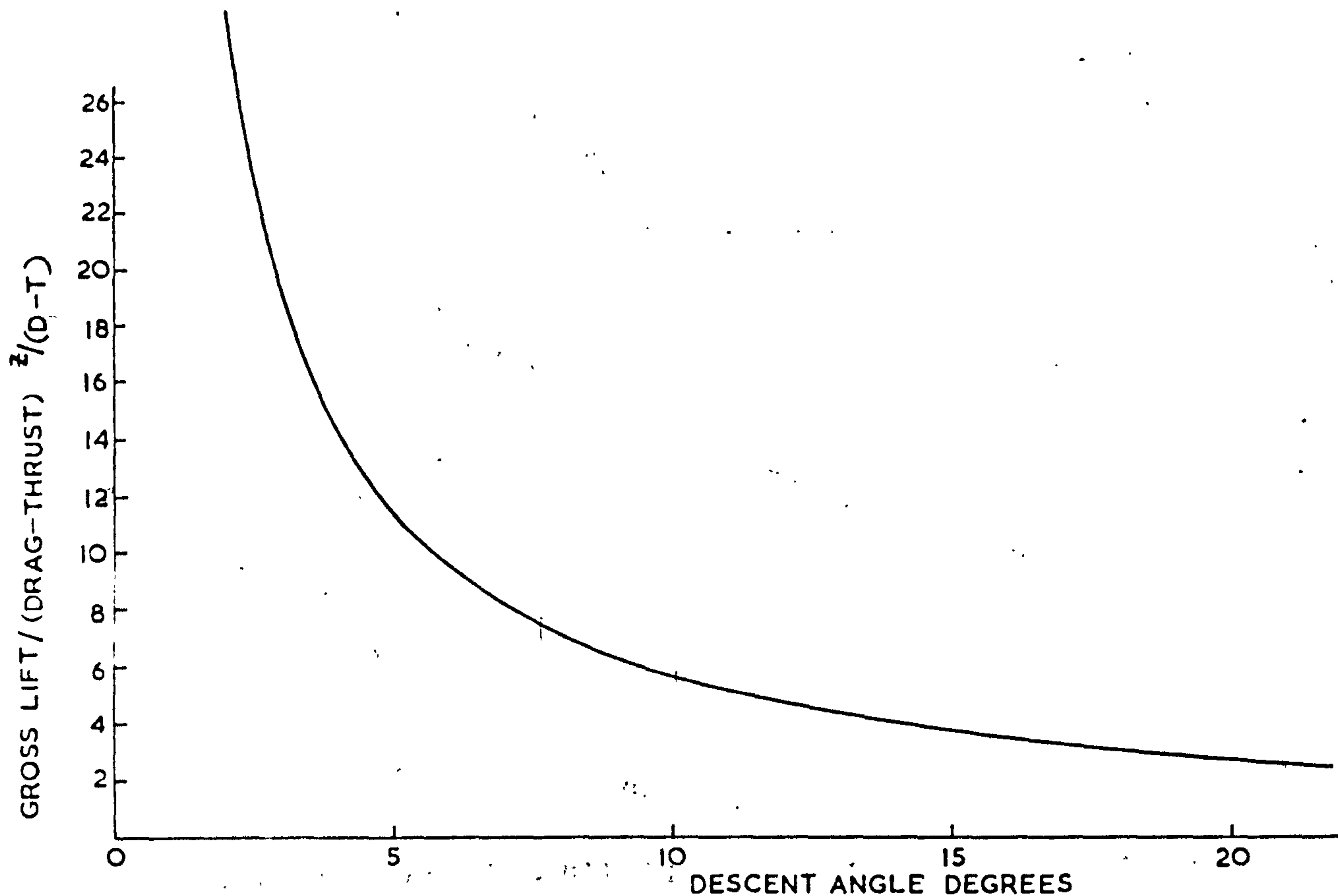


FIG. 4. AERODYNAMIC CHARACTERISTICS IN A STEADY DESCENT

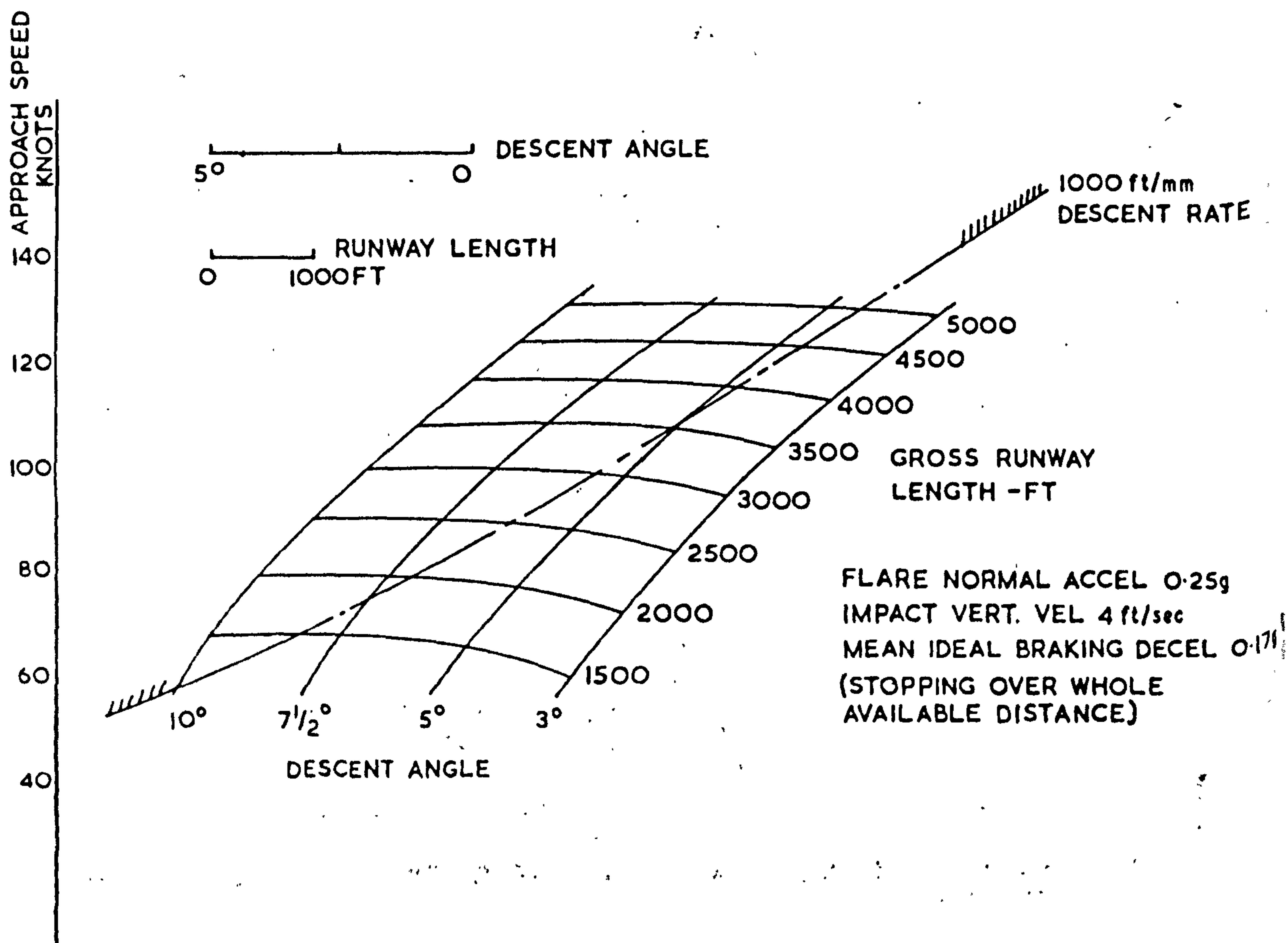


FIG. 5. APPROACH SPEED AS FUNCTION OF RUNWAY LENGTH AND APPROACH ANGLE

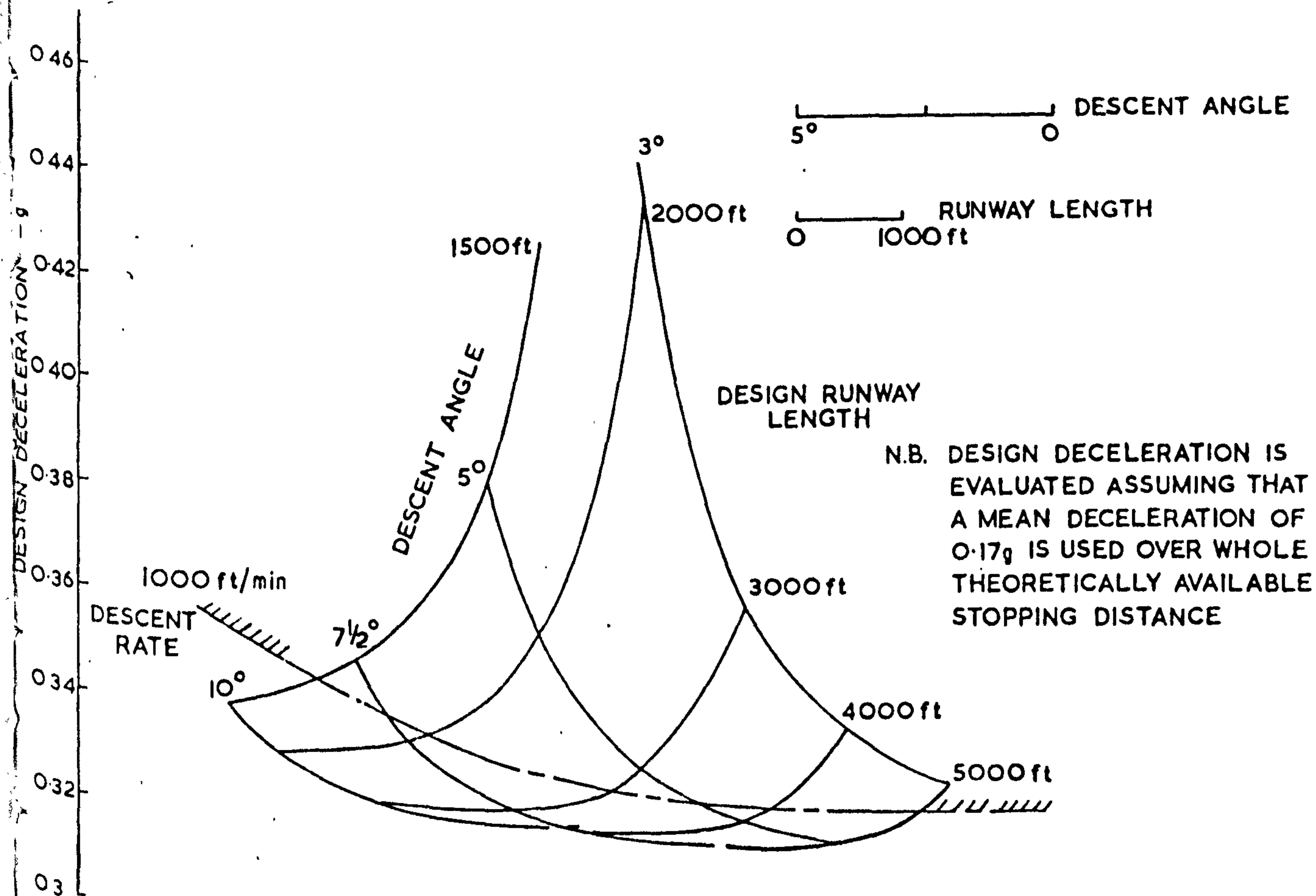


FIG. 6. EFFECTIVE MEAN DESIGN DECELERATION DURING BRAKING

7½° DESCENT ANGLE - INCOMPLETE
(FOR 3° DESCENT ADD 300FT APPROX)

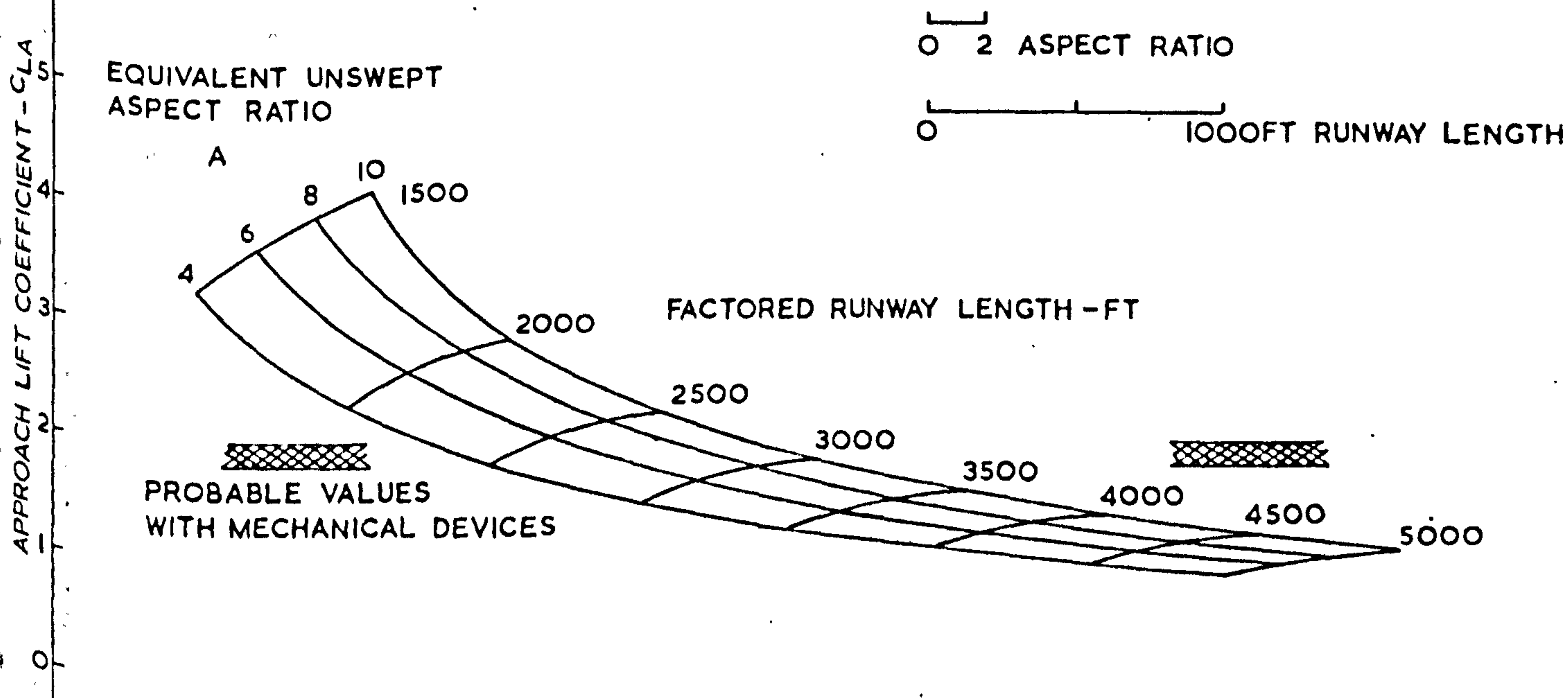


FIG. 7. REQUIRED APPROACH LIFT COEFFICIENT - 250 KNOTS EAS CRUISE

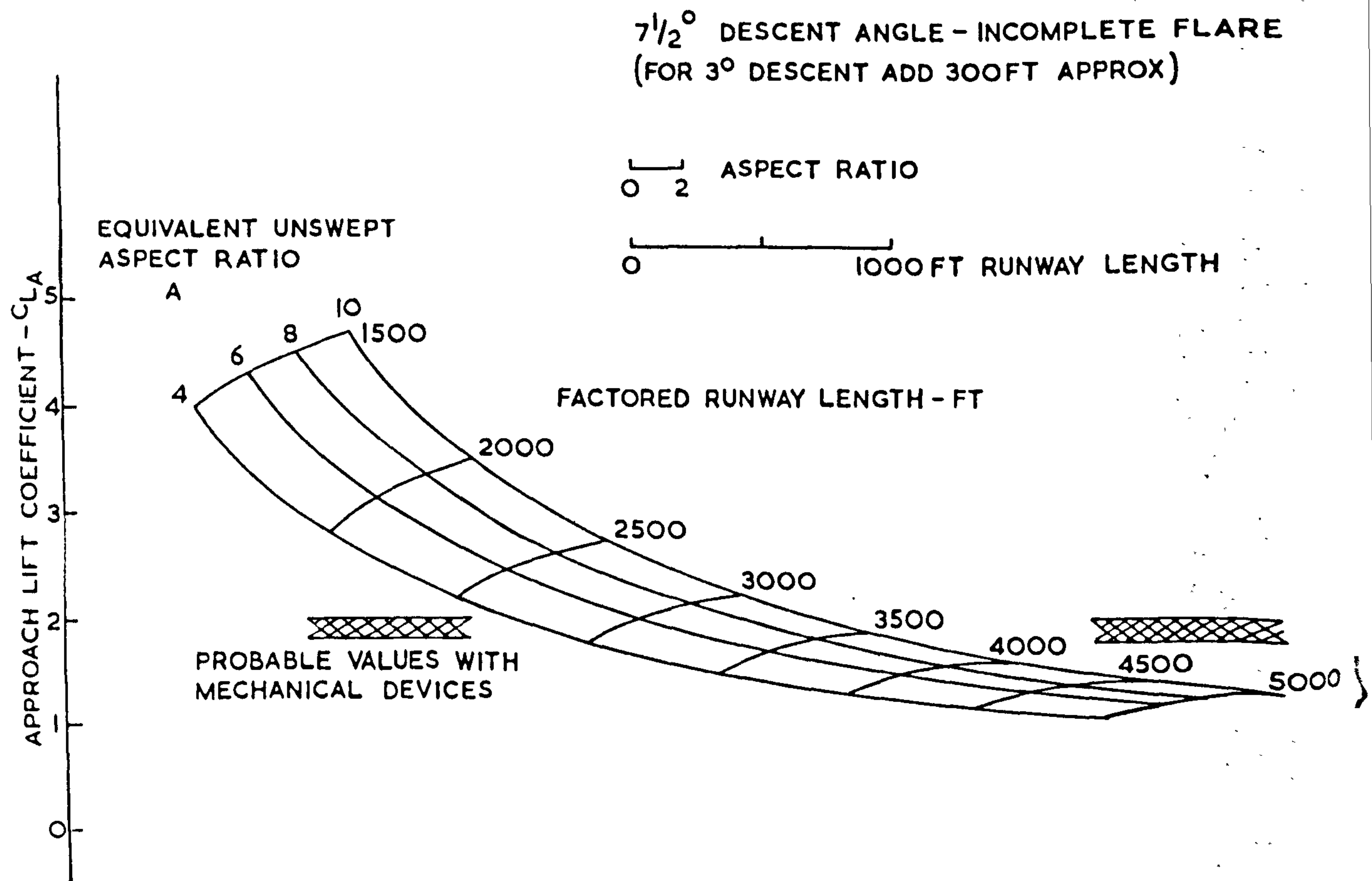


FIG. 8. REQUIRED APPROACH LIFT COEFFICIENT - 300 KNOTS EAS CRUISE

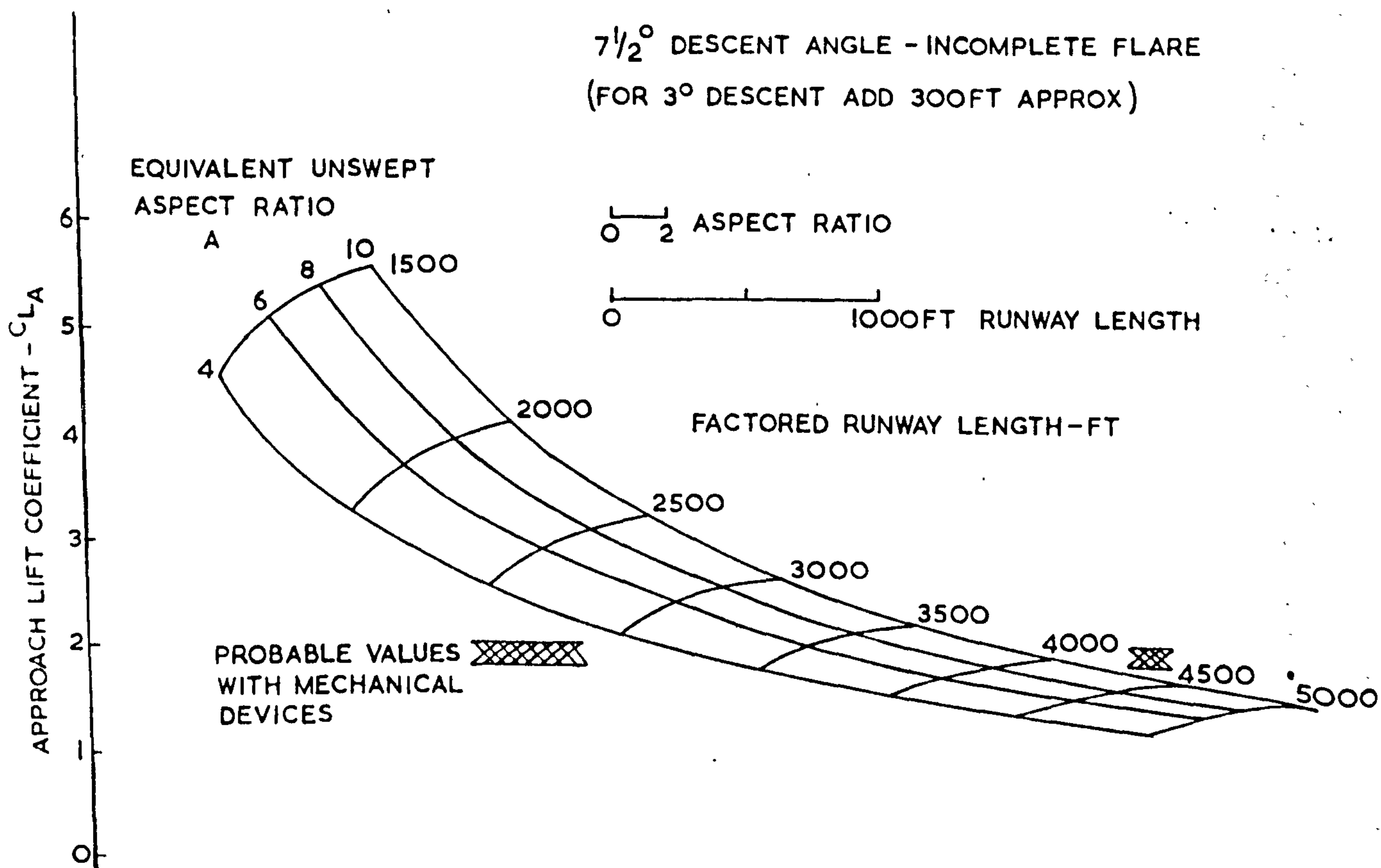


FIG. 9. REQUIRED APPROACH LIFT COEFFICIENT - 350 KNOTS EAS CRUISE

7½° DESCENT ANGLE
(FOR 3° DESCENT ADD 300FT APPROX)

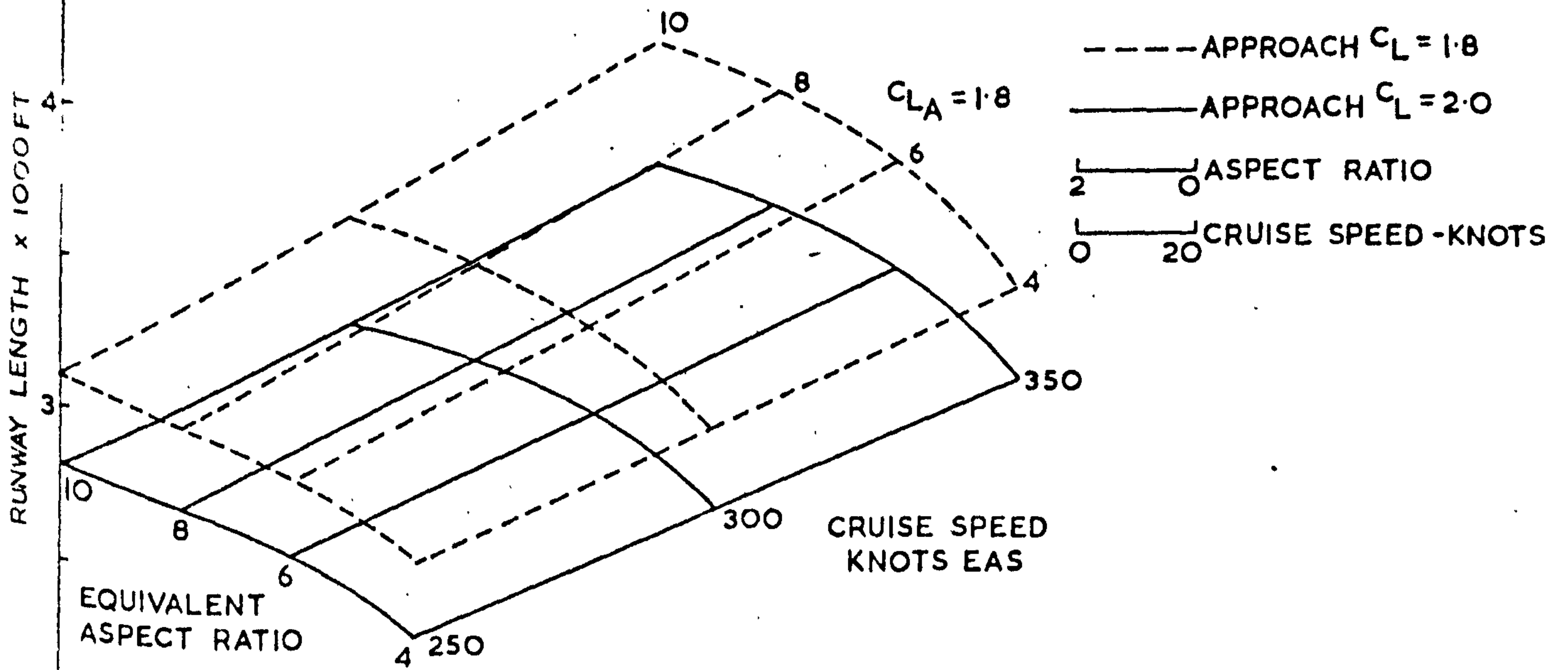


FIG. 10. LANDING RUNWAY CRITERIA – MECHANICAL HIGH LIFT DEVICES

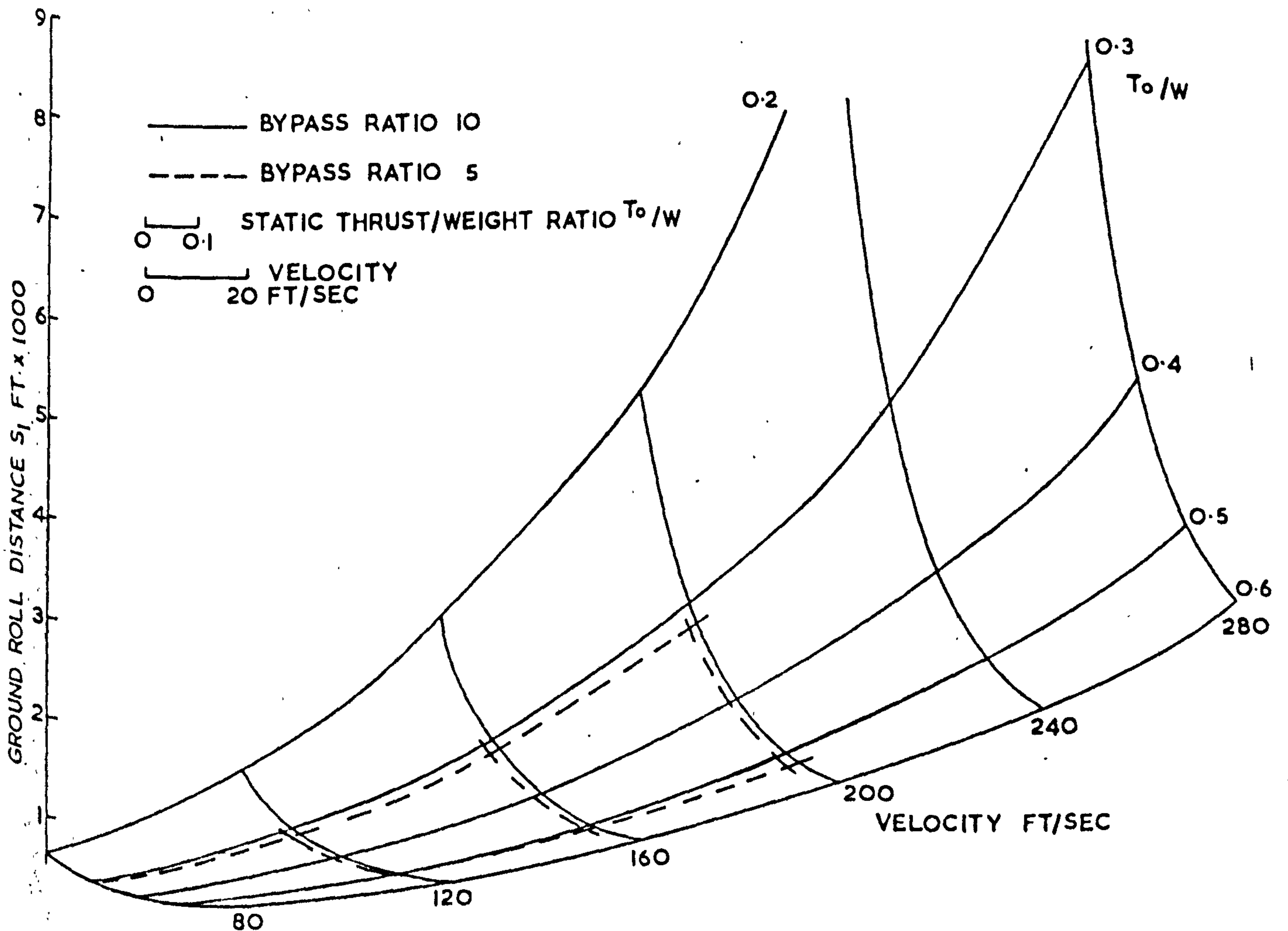


FIG. 11. GROUND ROLL DISTANCE AT TAKE OFF

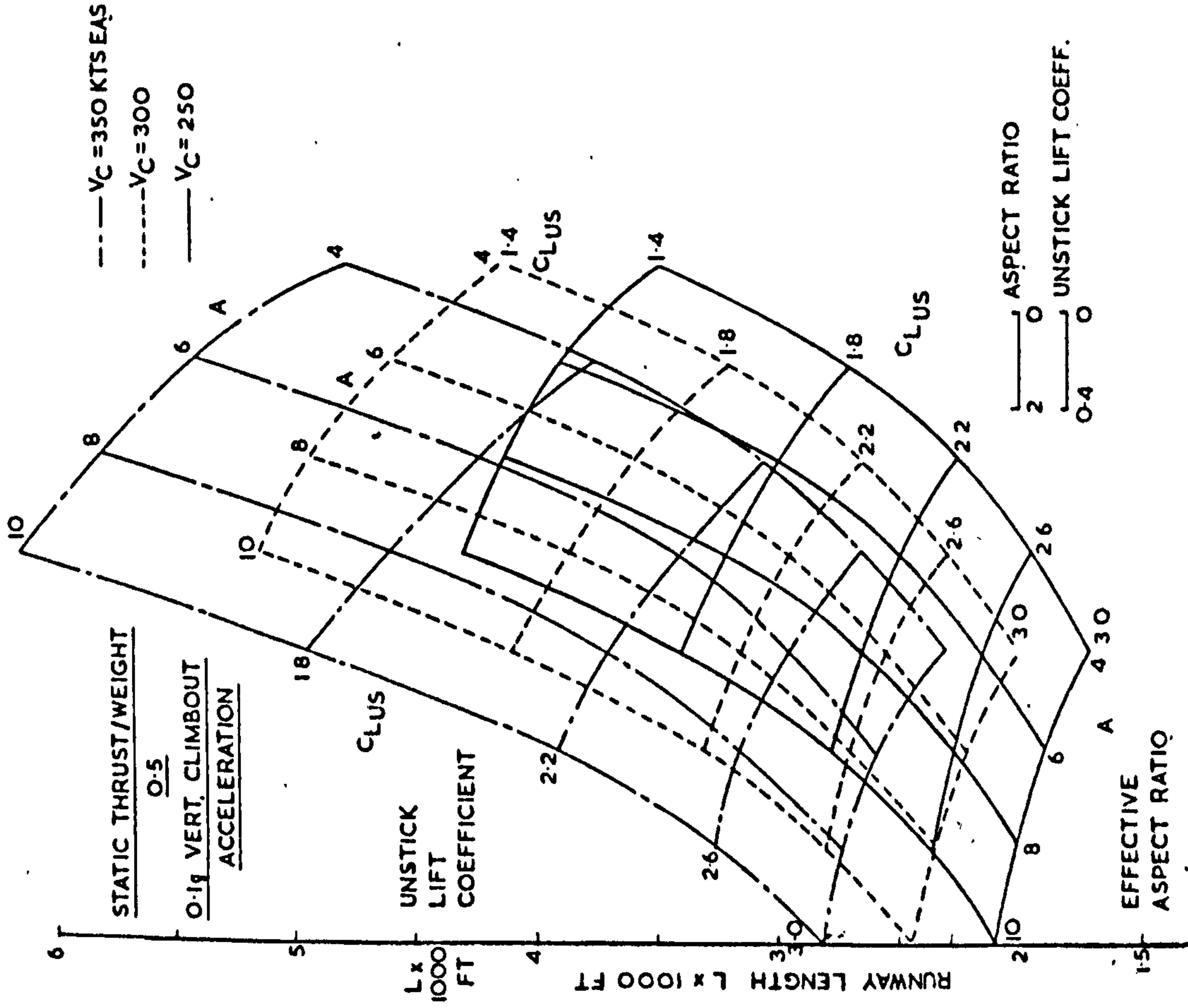


FIG. 14. TAKE OFF DISTANCE AS A FUNCTION OF CRUISE SPEED AND UNSTICK LIFT COEFFICIENT

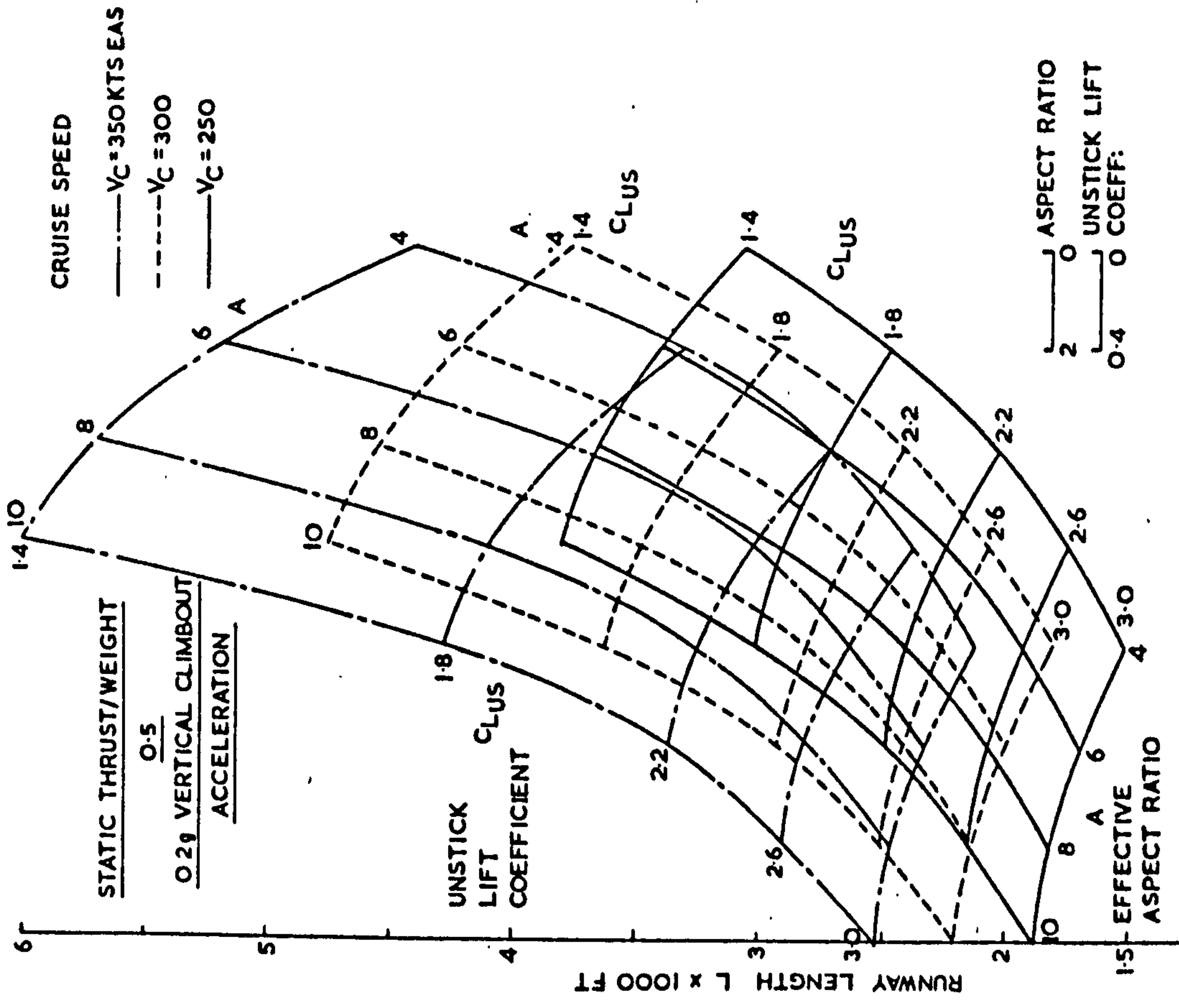


FIG. 15. TAKE OFF DISTANCE AS A FUNCTION OF CRUISE SPEED AND UNSTICK LIFT COEFFICIENT

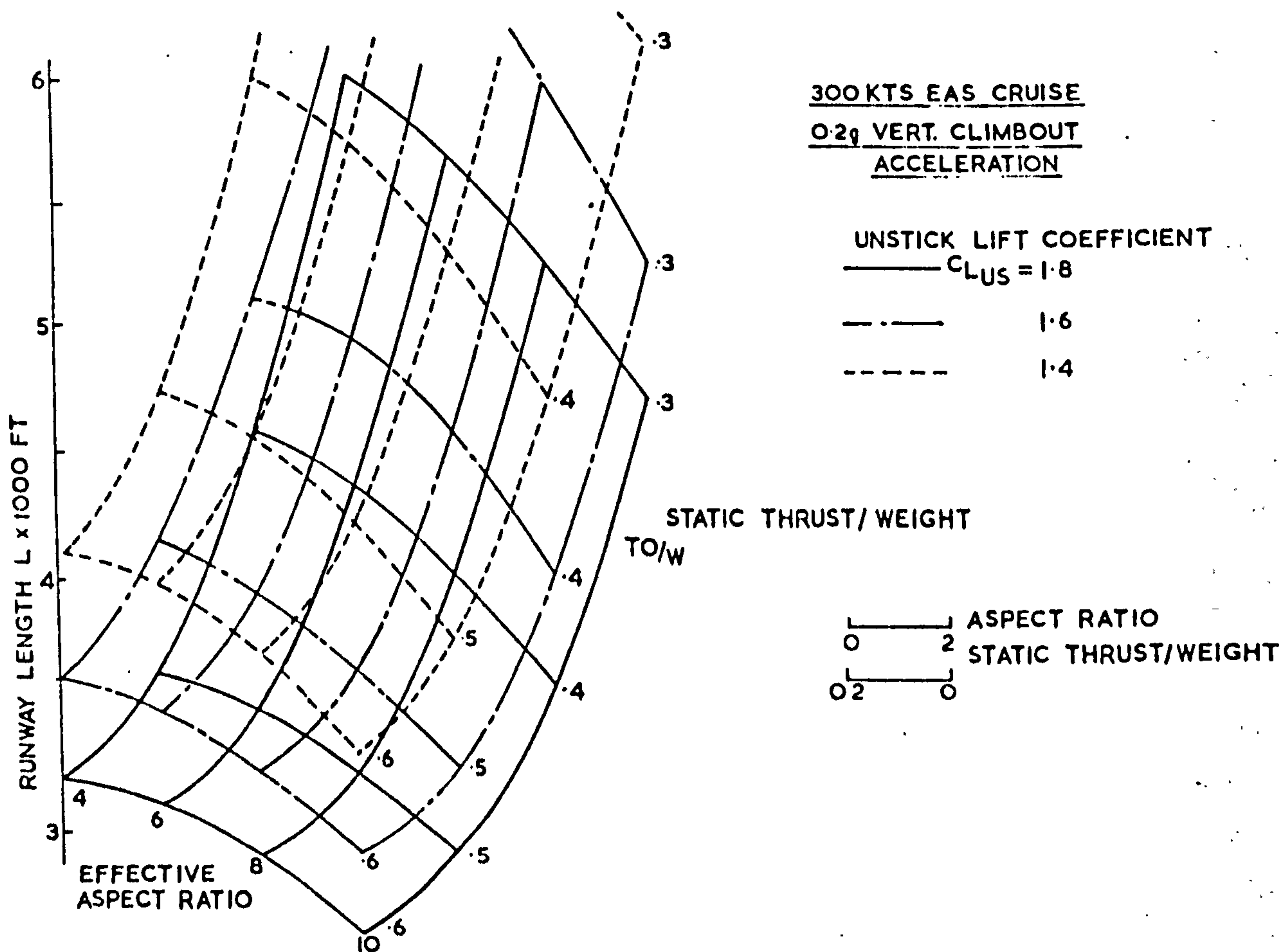


FIG. 16. TAKE OFF DISTANCE AS FUNCTION OF THRUST/WEIGHT AND UNSTICK LIFT COEFFICIENT

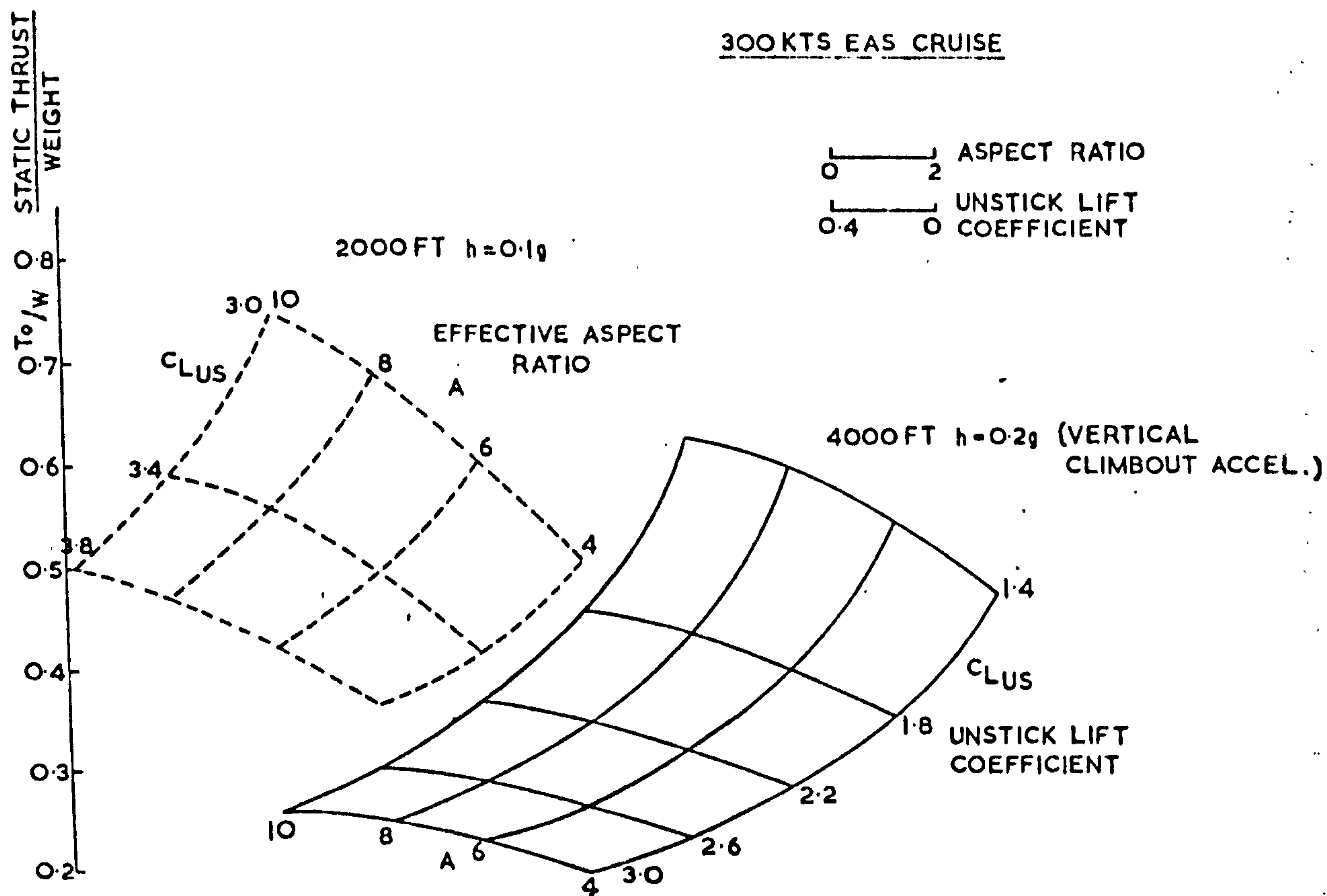


FIG. 17. THRUST/WEIGHT RATIO REQUIRED TO TAKE OFF FROM 2000 FT AND 4000 FT.

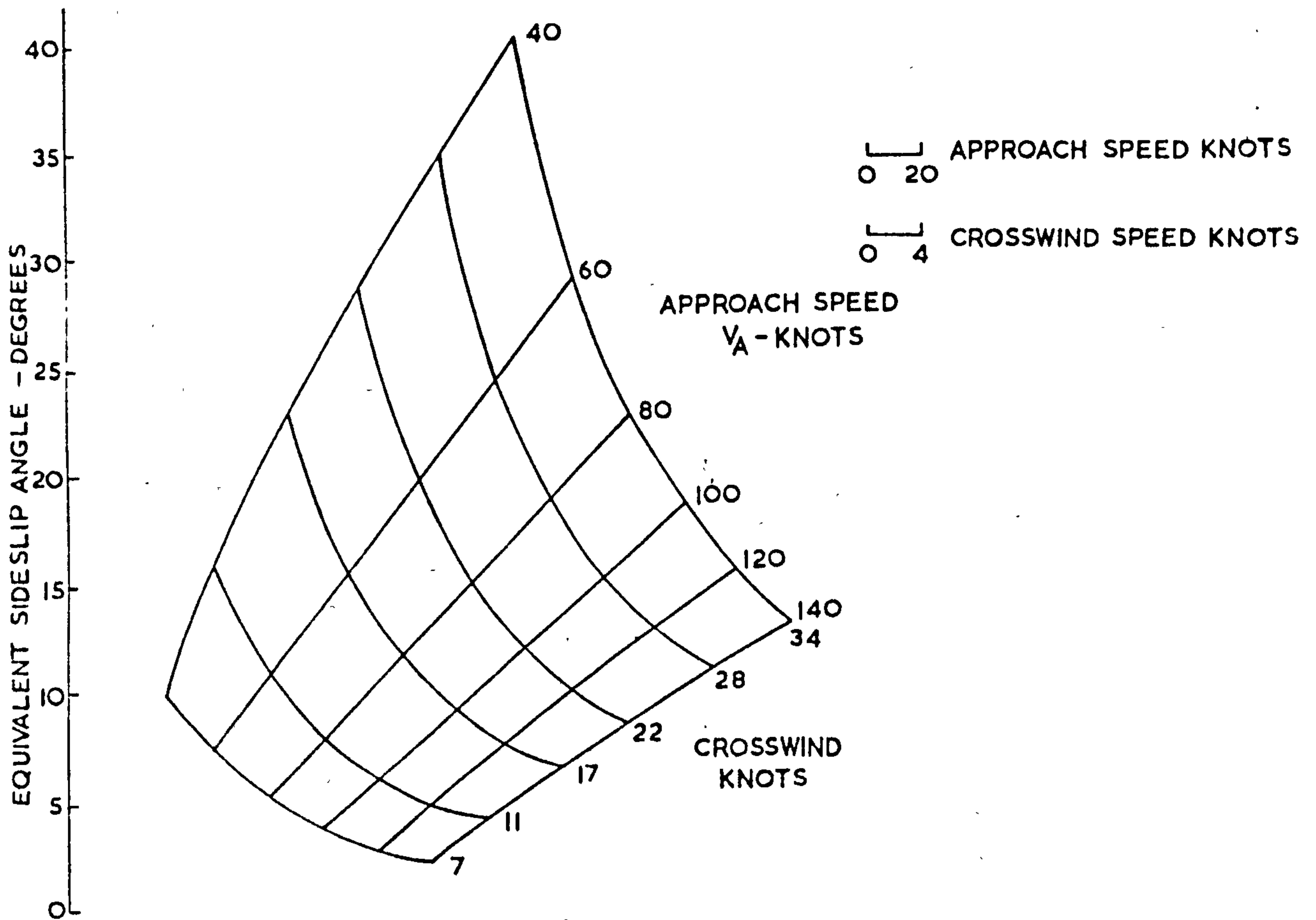


FIG. 18. EQUIVALENT SIDESLIP ANGLE IN CROSSWIND LANDINGS

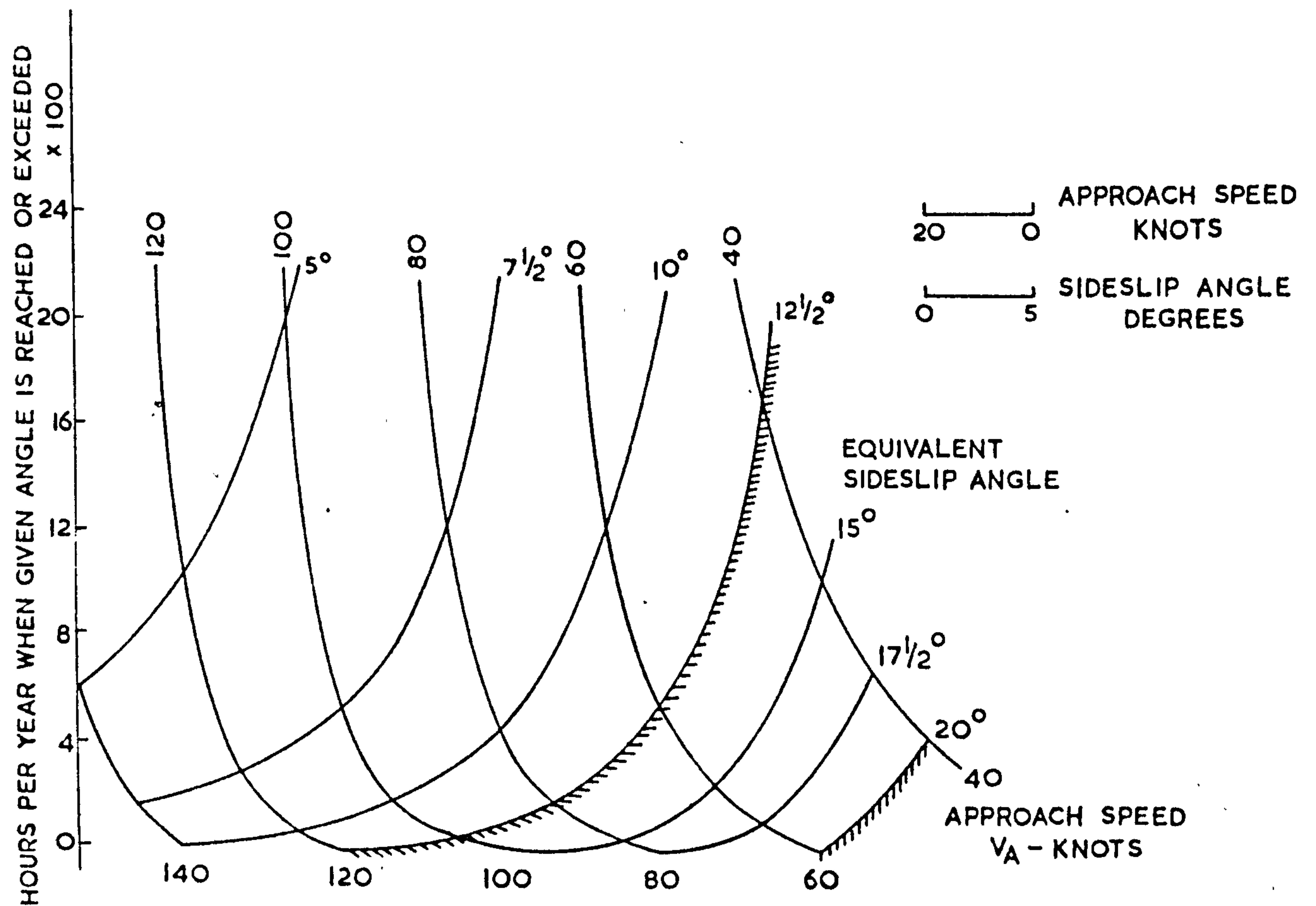


FIG. 19. ANNUAL FREQUENCY OF EQUIVALENT CROSS WIND SIDESLIP ANGLE AT CROYDON

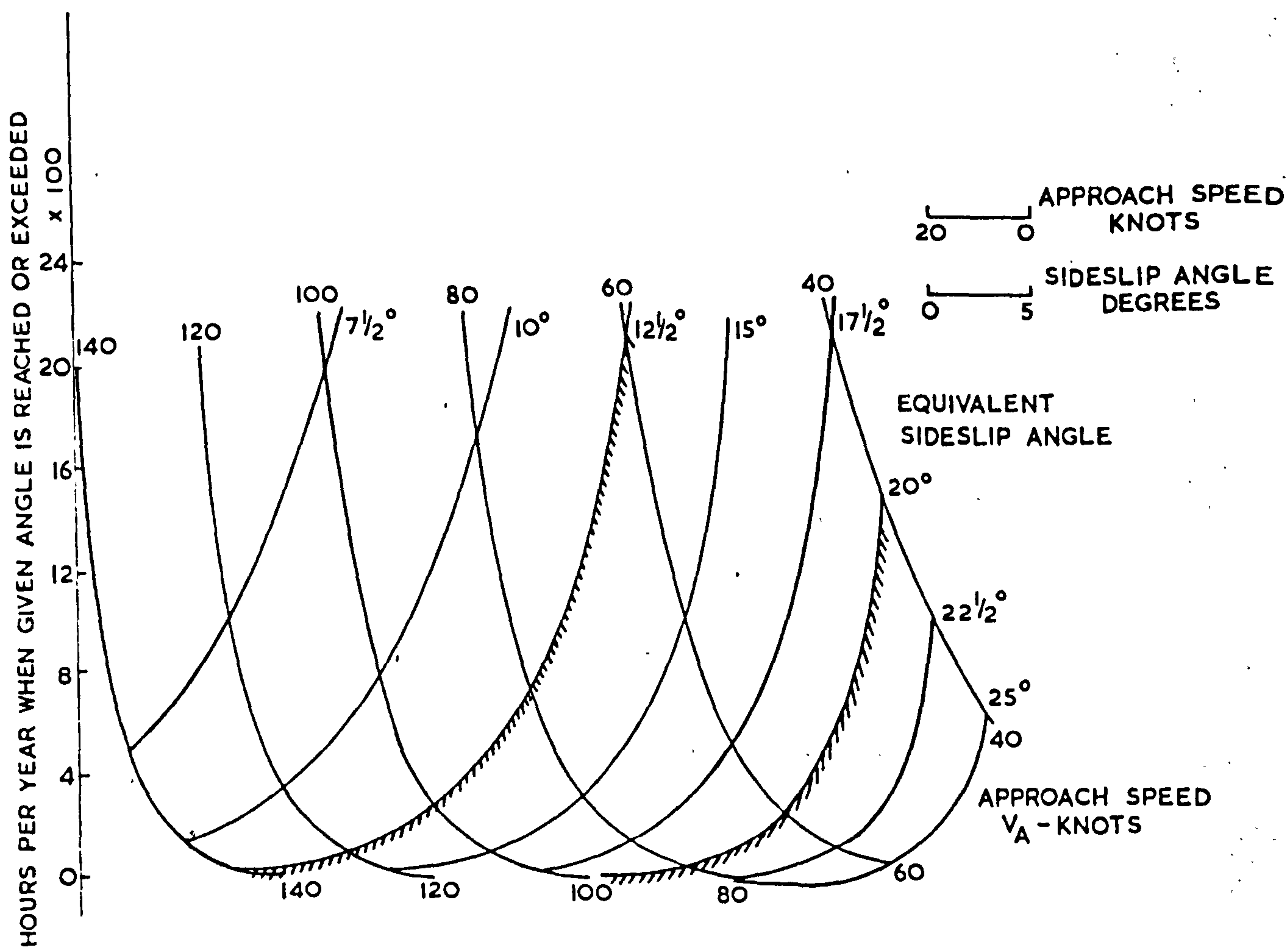


FIG. 20. ANNUAL FREQUENCY OF EQUIVALENT CROSS WIND SIDESLIP ANGLE AT SPEKE

Cranfield Report Aero No. 25

July 1974

CRANFIELD INSTITUTE OF TECHNOLOGY

College of Aeronautics

SUBSONIC JET TRANSPORT NOISE -
THE RELATIVE IMPORTANCE OF VARIOUS PARAMETERS

by

D. Howe

SUMMARY

The area of the 80 PNdB noise footprint of subsonic jet transport aircraft has been evaluated using a simple expression for powerplant noise level. The parameters varied were the bypass ratio, field length, climb out and descent angle, installed thrust, standard of engine acoustic treatment and the rate of noise attenuation. Curves are presented for typical ranges of the variables.

It was concluded that the bypass ratio is the most important influence on the footprint area. The attenuation rate also has a very significant effect but it is outside the control of the designer. Field length has only secondary effect.

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FIGURES

1. Variation of 80 PNdB Footprint with Thrust and Bypass Ratio. (Standard powerplants - shallow climb and approach) 6.3 dB attenuation.
2. As Figure 1, with steep climb and approach.
3. Variation of 80 PNdB Footprint with Thrust and Bypass Ratio. (Quietened powerplants - shallow climb and approach) 6.3 dB attenuation.
4. As Figure 3, with steep climb and approach.
5. As Figure 1, with 8 dB attenuation.
6. As Figure 2, with 8 dB attenuation.
7. As Figure 3, with 8 dB attenuation.
8. As Figure 4, with 8 dB attenuation.
9. Effect of Field Length on Footprint Area (Standard powerplants - bypass ratio 5)
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13. Effect of Attenuation on ratio of 80 PNdB and 90 PNdB Footprint Areas.
14. Effect of parameters in reducing footprint area relative to first generation long range transport aircraft characteristics.
15. Effect of parameters in reducing footprint area relative to wide body jet transport aircraft characteristics.

1. INTRODUCTION

Some of the earlier investigations into the operating characteristics of STOL and RTOL aircraft suggested that reduction in field length requirements is accompanied by a reduction of noise nuisance to the surrounding environment. More recent work, such as Ref. 1, has shown that this is not necessarily the case. Whilst there may be some slight reduction in noise nuisance with reduced field length the effect does not appear to be very large for the high bypass ratio powerplants envisaged for use in the next generation of subsonic jet transport aircraft. Indeed there appears to be little doubt that the noise nuisance becomes more severe when the reduction of field length is accompanied by the use of some form of powered lift.

The purpose of the present study is to investigate the relative importance of the main parameters governing the noise nuisance of aircraft which do not use powered lift, and to reconcile the apparent discrepancies between the earlier and more recent work on short and reduced field length designs.

2. BASIC CONSIDERATIONS

2.1 Noise nuisance criteria

A number of criteria have been established to compare the noise nuisance of aircraft. Basically these can be divided into two categories:-

- a) Evaluation of noise index contours, such as NNI, which are derived by weighting the noise levels of a given aircraft by the number of operations in a given time. Refinements of this approach, such as that suggested by Richards and Ollerhead (Ref.2) enable an assessment to be made on an overall basis so that the conditions at different airports may be compared.
- b) Establishing the ground noise footprint area outside of which the noise level falls below that of the general environment and there is thus no nuisance. In this case the frequency of operations has no relevance.

It is not intended to discuss all the relative advantages and disadvantages of the two approaches. The first is directly related to current certification procedures and operations. However the second is used for purposes of this comparative study because of its relative simplicity, independence of a particular airport and operational pattern, and especially because it is likely that it will be the ultimate requirement.

The acceptable background noise level in a given environment varies according to circumstances and has not yet been completely defined. In the case of an urban environment the level is probably in the 66 dBA to 70 dBA range (Ref.3,4) although recent work by Lilley (Ref.5) suggests a somewhat lower figure of 55 to 65 dBA. Aircraft noise subjectively adds from 10 to 14 dB and therefore the higher of the above suggestions implies that the 80 PNdB noise footprint is a suitable basis for comparison. Although this may be somewhat high on the basis of Lilley's findings it is convenient to use and is the criteria adopted for comparison in this investigation. In the past the 90 PNdB footprint area was frequently used for comparison purposes.

2.2 Aircraft noise source

There is some evidence to suggest that airframe noise may be significant in those cases when an aircraft is powered by very quiet fan engines (Ref.1). This effect is very dependent upon the forward speed and whilst it may become the dominating factor in some cases, for simplicity it is neglected in the present comparison. The noise of a turbojet results from three basic elements and the interactions between them, namely the fan/compressor turbine and exhaust. A very important parameter is the specific thrust, (Ref.6) that is the ratio of the static thrust to the total mass flow through the unit. Taken with the core engine thermodynamic cycle it defines the bypass ratio. Published data for engines designed at the current level of technology and with typical core cycles suggests (Ref. 1):-

$$\text{PNdB} = 10\log \left[\text{Antilog}(8.8 + \log R) + \text{Antilog}(11.5 - 1.7\log R) + \text{Antilog}(12.8 - 3\log R) \right] + 10\log \left(\frac{T}{80000} \right) - \Delta \dots (1)$$

at 500 ft distance from the aircraft.

where R is the bypass ratio
T is the static thrust in lbs
Δ is a correction which allows for the standard of acoustic treatment employed, extent of noise shielding achieved, etc. Present evidence suggests that Δ may eventually exceed 10 dB for engines of high bypass ratio.

A simple formula such as Eq.(1) can only be regarded as being very approximate since it makes no allowance for directional and other important effects. It cannot be used to predict the effective perceived noise level (EPNdB).

When the powerplants are operated at less than the static design thrust it can be assumed that the noise is reduced by:-

$$15 \left(\frac{T_R - T}{T} \right) \text{ dB} \quad \dots (2)$$

where T_R is the reduced thrust level.

2.3 Attenuation

The rate of noise attenuation is a very important parameter in the establishing of the footprint area. The theoretical inverse square law gives a 6 dB reduction in noise level for each doubling of the distance from the source. In free air conditions the rate of attenuation is normally taken as 6.3 dB/doubling of distance. Near to the ground the situation is more complex and is inevitably dependent upon local conditions. Measurements on rotorcraft noise suggest that the decay rate may be as high as 8 dB for air to air and 10 dB for ground to ground conditions (Ref.7). Such high attenuation rates cannot be relied upon in practice and it is probable that absolute distance influences the situation as well as relative values. Near to the ground an attenuation of about 7 dB/doubling of distance is probably realistic.

An indication of the importance of this parameter in determining footprint area can be seen by reference to Figure 13. This shows the ratio of the 80 PNdB to 90 PNdB footprint areas as a function of the attenuation rate beyond the 500 ft datum distance assumed in Eq.(1). The area ratio is about 9 for the 6.3 dB decay rate but it falls to about 7 for a 7 dB decay rate and 4 for the 10 dB rate.

2.4 Evaluation of Footprint Area

The footprint area can be evaluated using Eqs.(1) and (2) in conjunction with an assumed attenuation rate. In the present case a very simple representation of the aircraft flight path was assumed.

- a) The take off footprint area was calculated by assuming a rectangular shape defined in width by Eq.(1) and the attenuation to 80 PNdB and in length by the assumed field length.
- b) Climb out footprint area was taken as the elliptical shape obtained by assuming a given climb angle and the same width as that used for the take off phase. Thus no allowance was made for any noise abatement procedures or change in attenuation rate with increased altitude.
- c) The landing footprint area was evaluated in an exactly similar way using the assumed descent angle and a width based on a reduced thrust appropriate to landing conditions. For simplicity this was assumed to be 33% of the static value in all cases which implies a 10 dB reduction in the noise level at 500 ft distance relative to the value given by Eq.(1).

2.5 Parametric Variations

The 80 PNdB footprint area was calculated for the following variations of parameters:-

- a) Bypass ratios of 1, 5 and 10.
- b) Total static thrust of 40,000 lb, 80,000 lb and 160,000 lb. These values were chosen to be representative of the range likely to be found in transport aircraft varying from the smaller short haul RTOL concepts to long range large subsonic aircraft.
- c) Runway field lengths of 4000 ft, 7000 ft and 10,000 ft.
- d) Attenuation rates of 6.3 dB, 8 dB and 10 dB per doubling of distance from the 500 ft datum.
- e) Two engine acoustic treatment conditions:-
 $\Delta = 0$ to represent a 'standard' powerplant except that a 5 dB reduction was allowed for an exhaust silencer on the take off conditions of the bypass ratio one case only.
 $\Delta = 5$ dB to represent a 'quietened' powerplant in all cases.
- f) Shallow and steep climb out and approach conditions
These were defined as:-
Shallow climb at 10° and shallow descent at 3° .
Steep climb at 20° and steep descent at 6° .

3. RESULTS

The results of the calculations are presented in carpet form in Figures 1 to 8. These give the variation of the 80 PNdB footprint area as a function of two basic parameters, namely the bypass ratio and total static thrust, for the various combinations of the other parameters. It was found to be necessary to use a logarithmic scale for the footprint area to show meaningful values for the higher bypass ratio cases. Figures 1 to 4 are based on the 6.3 dB attenuation rate whilst Figures 5 to 8 cover the 8 dB condition. In each set the first two figures refer to the standard noise condition and the latter two to the quietened powerplant. The first figure of each pair shows the results for the shallow climb and descent pattern whilst the second gives the corresponding information for the steep climb and descent.

Figures 9 and 10 extract the field length data as a function of thrust and climb/descent pattern for the particular cases of standard noise levels and bypass ratios of 5 and 10 respectively.

The effect of attenuation rate on footprint area is shown in Figures 11 and 12 as a function of thrust and runway length for two particular cases. These are a bypass ratio of 5 with shallow climb and descent and a bypass ratio of 10 with steep climb and descent. Standard noise conditions were assumed in both cases.

4. DISCUSSION OF RESULTS

The figures show clearly that the bypass ratio is by far the most important parameter in determining the noise footprint area for a given installed thrust. There is a very large reduction in the footprint area when the bypass ratio is increased in the range of one to five. The rate of reduction is appreciably less for further increase of bypass ratio but up to a value of ten it is nevertheless significant. It should be noted, however, that an increase of bypass ratio implies an increase in installed thrust for a given cruise performance so that the full effect of footprint area reduction may not be achieved in spite of the improved low speed characteristics.

Figures 11 and 12 demonstrate the importance of the rate at which noise is attenuated. This is, of course, outside the control of both the designer and operator. Here the point to be made is that it is essential to know the value assumed in any prediction made, otherwise comparisons are likely to be quite meaningless.

The effect of employing steep climb out and descent techniques can be seen by reference to Figure 9 and 10. A significant reduction in footprint area can be made. For example at the higher bypass ratio the 80 PNdB footprint area is reduced by some 30% for the lower installed thrusts and this increases to about 50% for higher thrust conditions. These reductions are equivalent to a bypass ratio increase of about two in the five to ten range.

These same figures also show that the runway length has only a secondary effect. Whilst there is some reduction in footprint area as field length is reduced the maximum improvement is only about 30% for a reduction from 10,000 ft to 4,000 ft. Since this shorter field length is also likely to imply increased installed thrust the practical gains may be expected to be much less. It would appear that the results obtained from early studies of STOL and RTOL aircraft were incorrect in anticipating significant footprint area reductions due simply to reduction of field length. The effect noted was due primarily to the higher bypass ratio powerplants proposed for these classes of aircraft, and to the steeper climb and descent paths anticipated.

Acoustic treatment of the powerplants to reduce the basic noise level by the datum value of 5 dB assumed for the 'quietened' aircraft has the greatest effect at high bypass ratios and low installed thrust. In this case the reduction of the 80 PNdB footprint area varies from as much as 65% for an attenuation rate of 6.3 dB/doubling of the distance to about 50% when it is 8 dB. At lower bypass ratios and higher installed thrusts the absolute reduction in footprint area is significant even though the relative values are less.

Figures 14 and 15 illustrate the relative importance of the parameters studied in the context of two particular reference conditions. The former of these shows the reduction in the noise footprint area when the parameters are varied over a practical range from a condition representative of that of the first generation of long range jet transport aircraft. As would be expected the bypass ratio change is by far the most significant. The second of these figures uses the current generation of wide body long range transports as the reference with a datum bypass ratio of five. In this case the effect of the parameters is of the same order except for that of field length and to some extent the climb/descent pattern.

5. CONCLUSIONS

5.1 The field length capability of an aircraft has only a secondary effect on the noise footprint area.

5.2 The most important parameter within the control of the designer is the specific thrust of the powerplant. For a given core engine technology this parameter can be identified with bypass ratio. It would appear that bypass ratios in excess of ten yield only relatively small gains.

5.3 Steep climb out and approach techniques can have a significant effect in reducing the footprint area. Doubling the climb and descent angles relative to those typical for conventional aircraft is approximately equivalent to increasing bypass ratio by two in the five to ten range.

5.4 A 5 dB reduction in the basic noise of the powerplant achieved by acoustic treatment has a similar reduction in footprint area as that of doubling the climb and descent angle at low bypass ratio.

5.5 The noise attenuation rate is also of great importance but it is outside the control of the designer.

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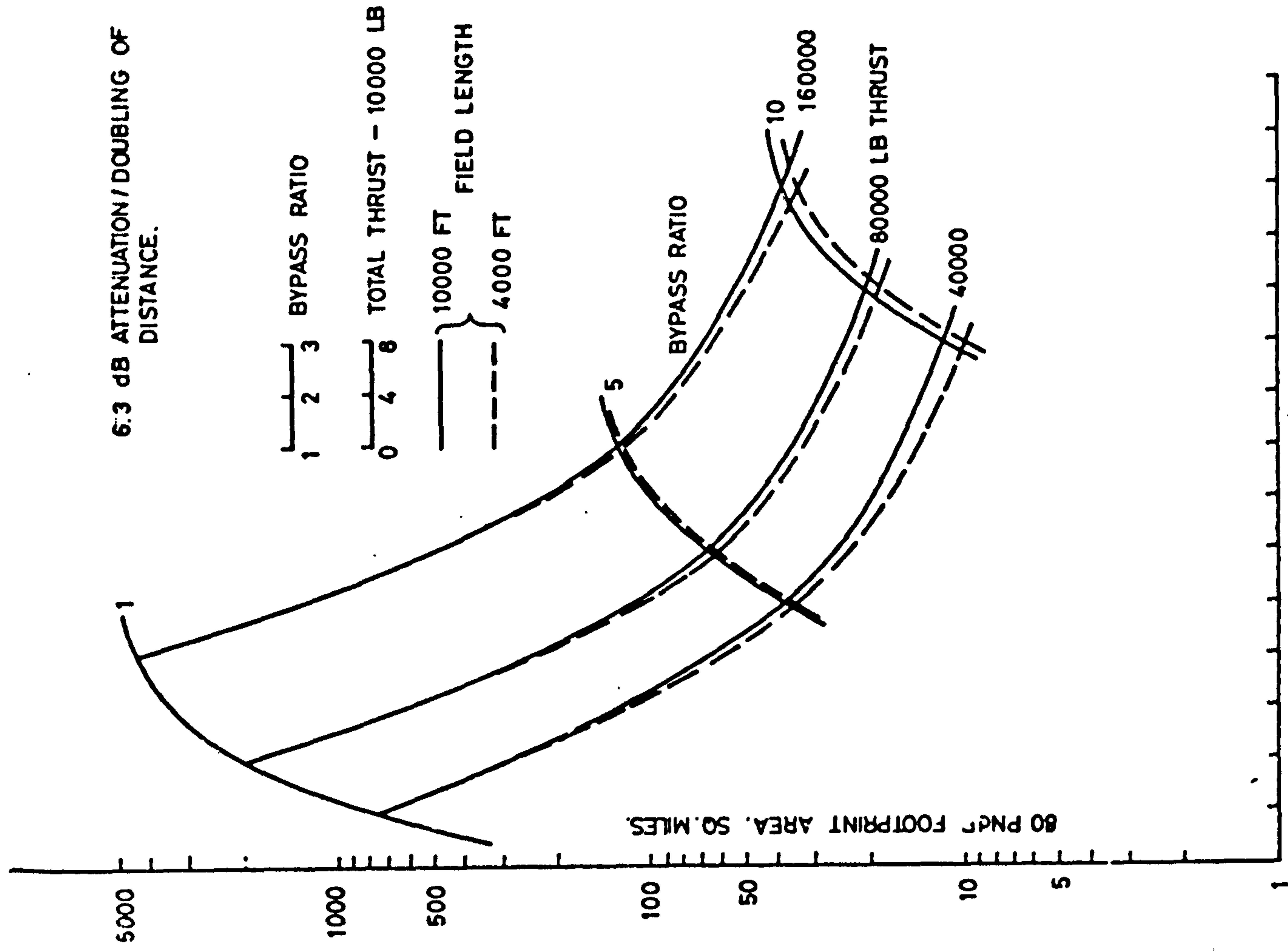


FIG. 1. VARIATION OF 80 PNdB FOOTPRINT WITH THRUST AND BYPASS RATIO. (STANDARD POWERPLANTS - SHALLOW CLIMB AND APPROACH) 6.3 dB ATTENUATION.

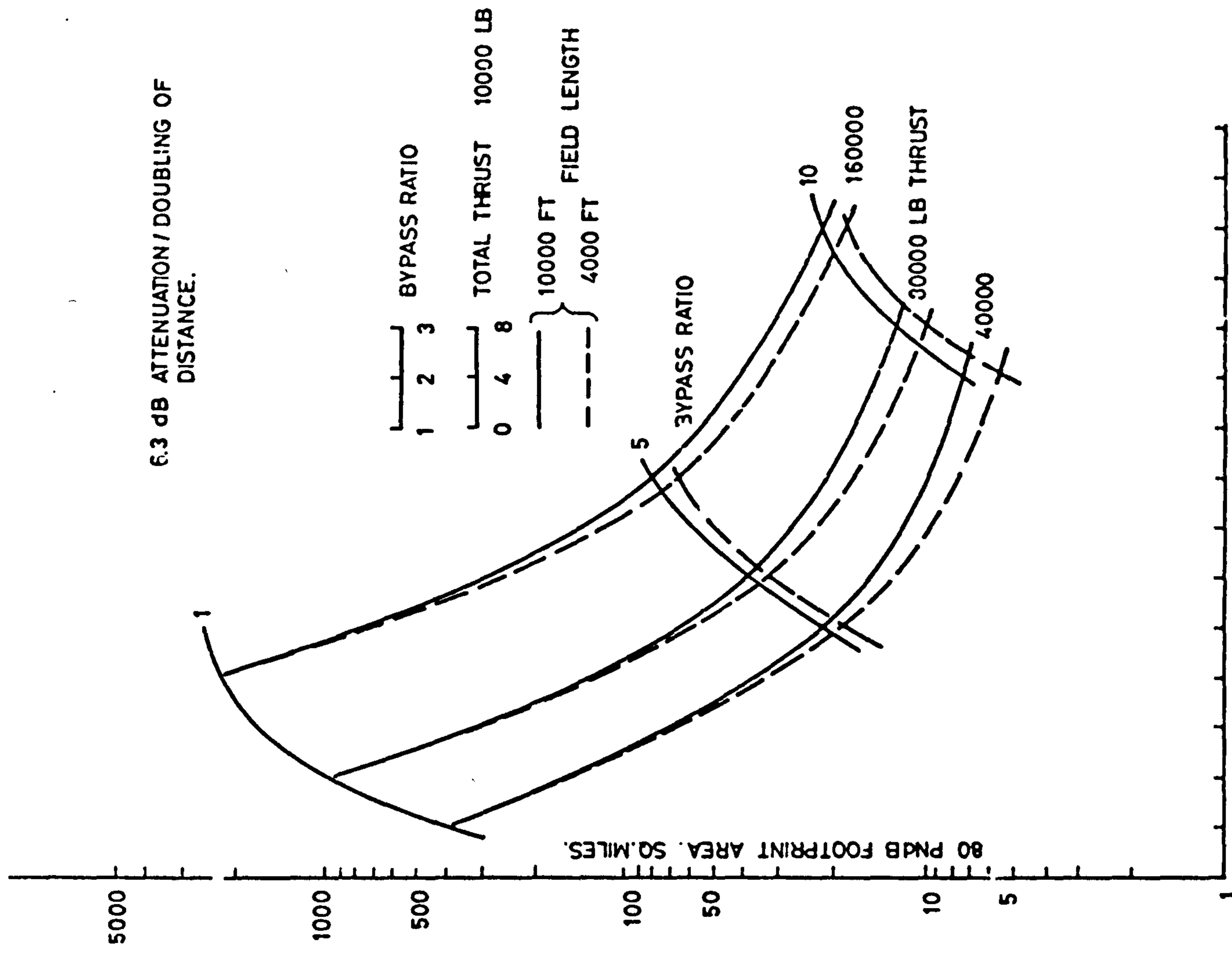


FIG. 2. AS FIGURE 1, WITH STEEP CLIMB AND APPROACH.

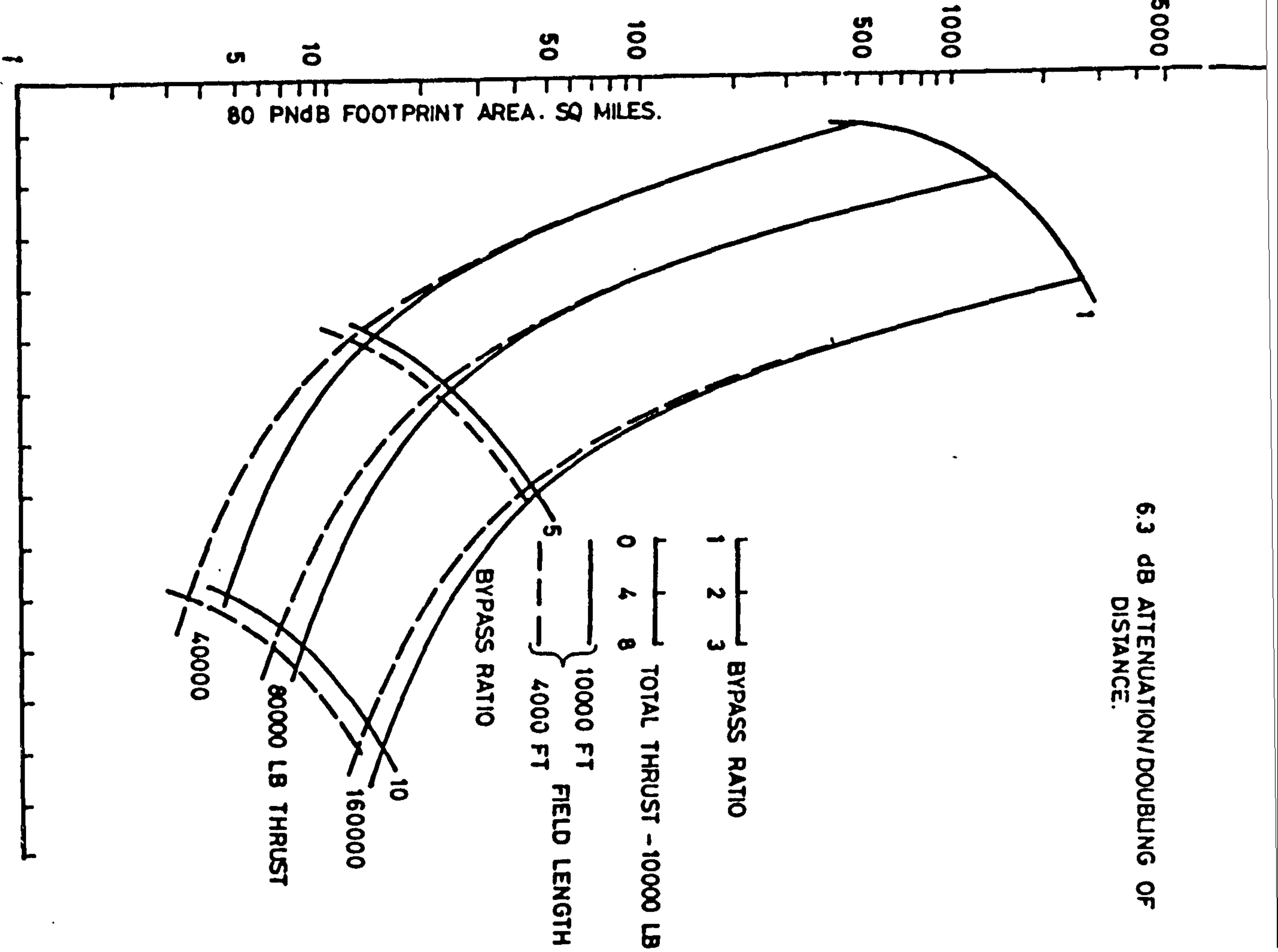


FIG. 3. VARIATION OF 80 PNdB FOOTPRINT AREA WITH THRUST AND BYPASS RATIO. (QUIETENED POWERPLANTS - SHALLOW CLIMB AND APPROACH) 6.3 dB ATTENUATION.

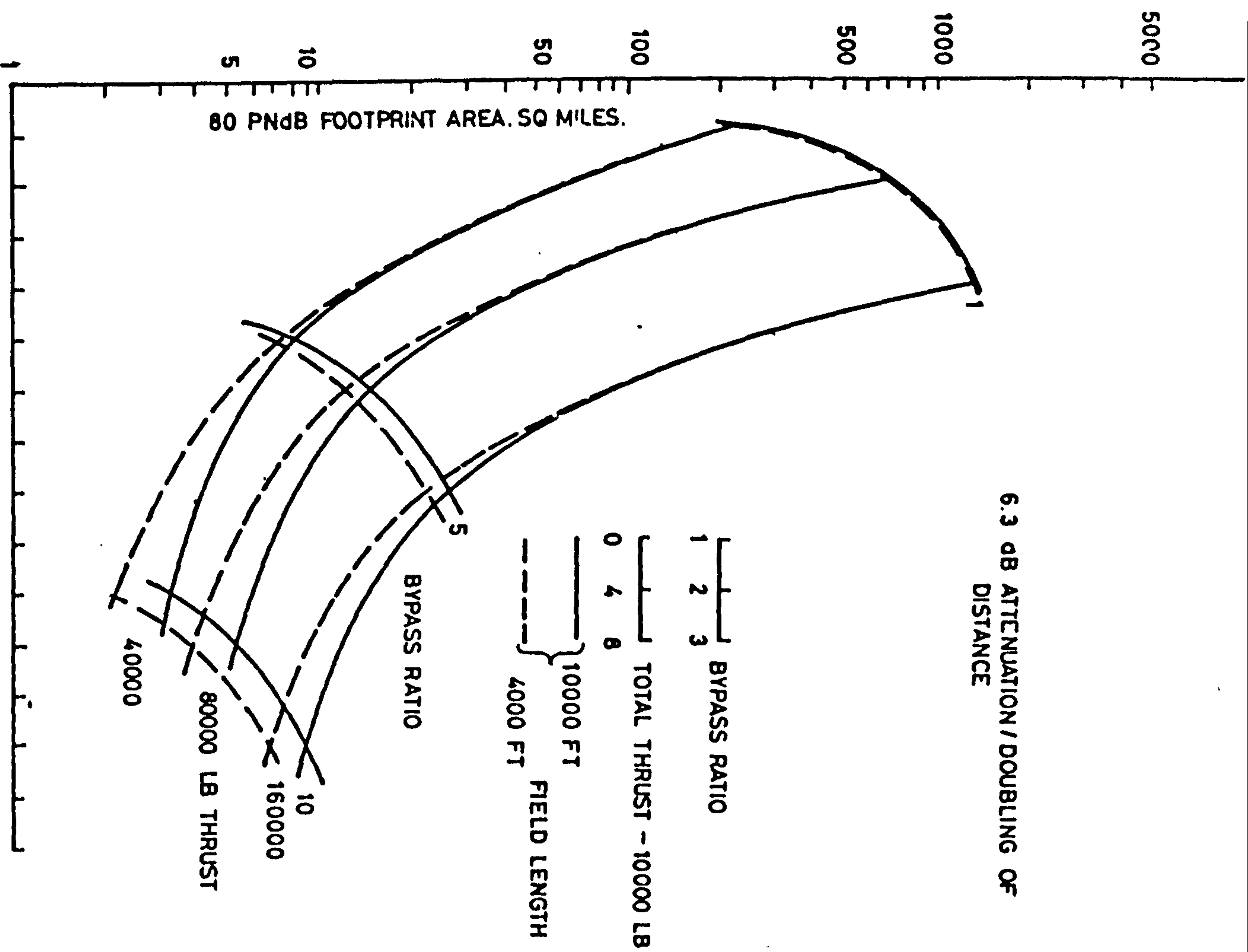


FIG. 4. AS FIGURE 3, WITH STEEP CLIMB AND APPROACH.

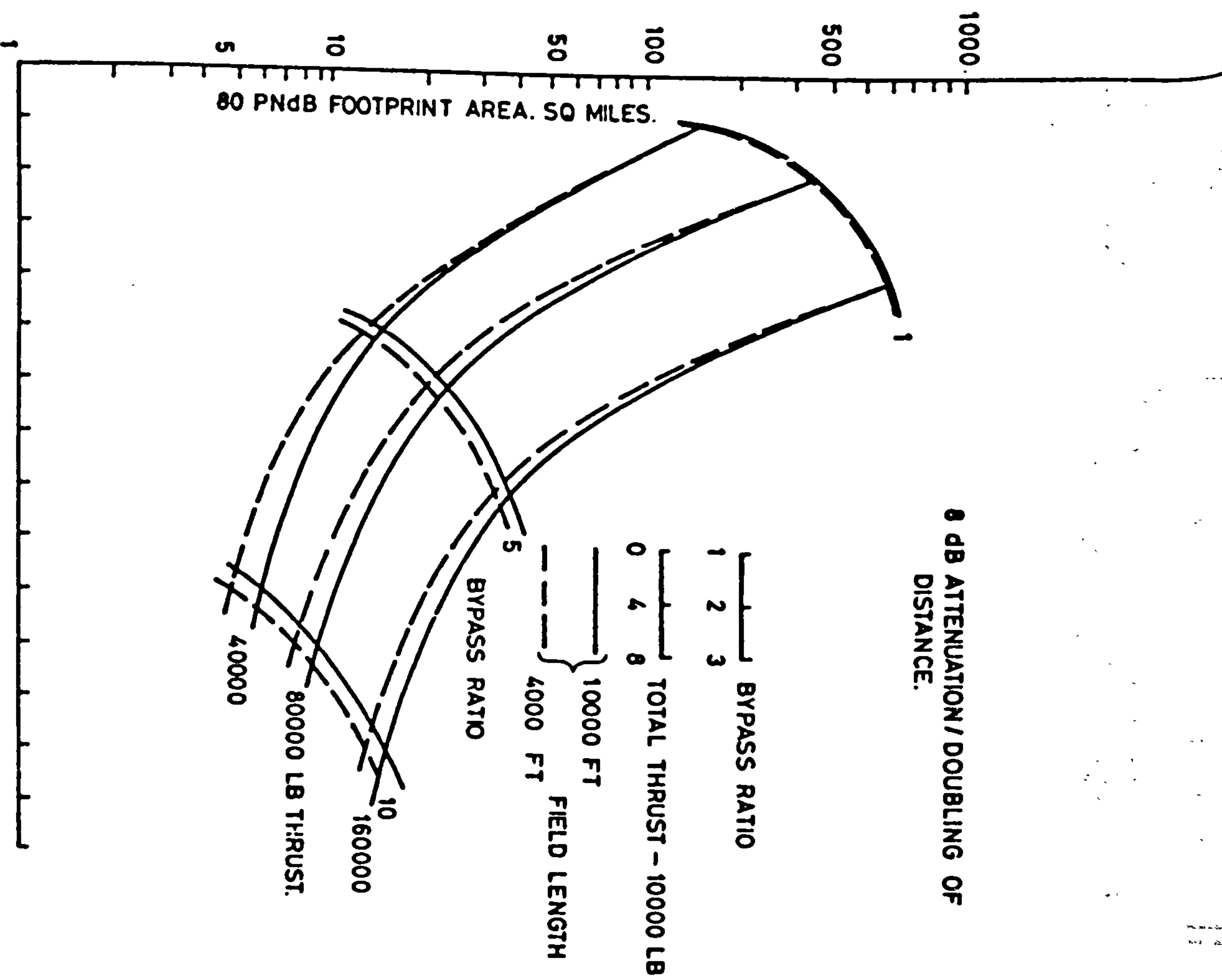


FIG. 5. AS FIGURE 1, WITH 8 dB ATTENUATION.

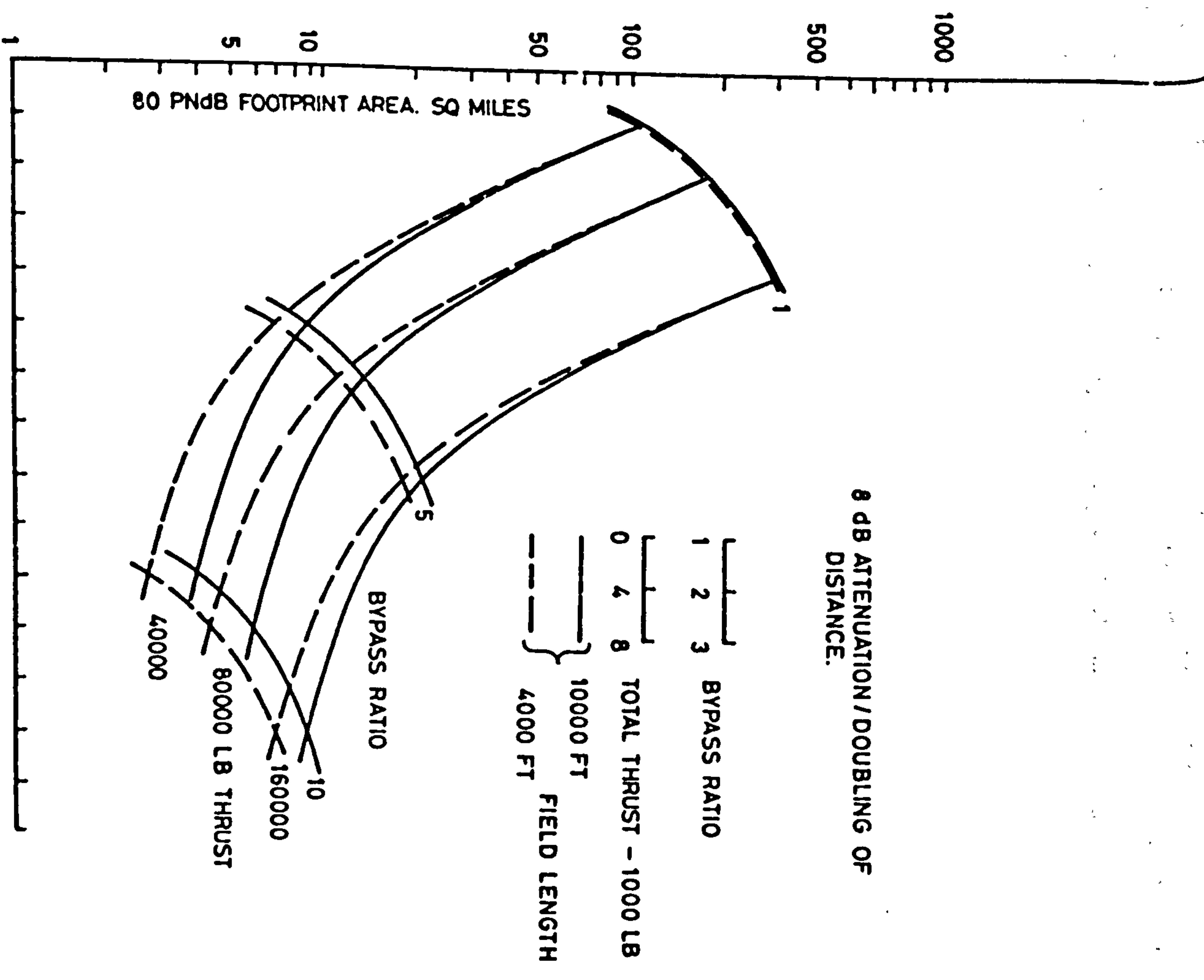


FIG. 6. AS FIGURE 2, WITH 8 dB ATTENUATION.

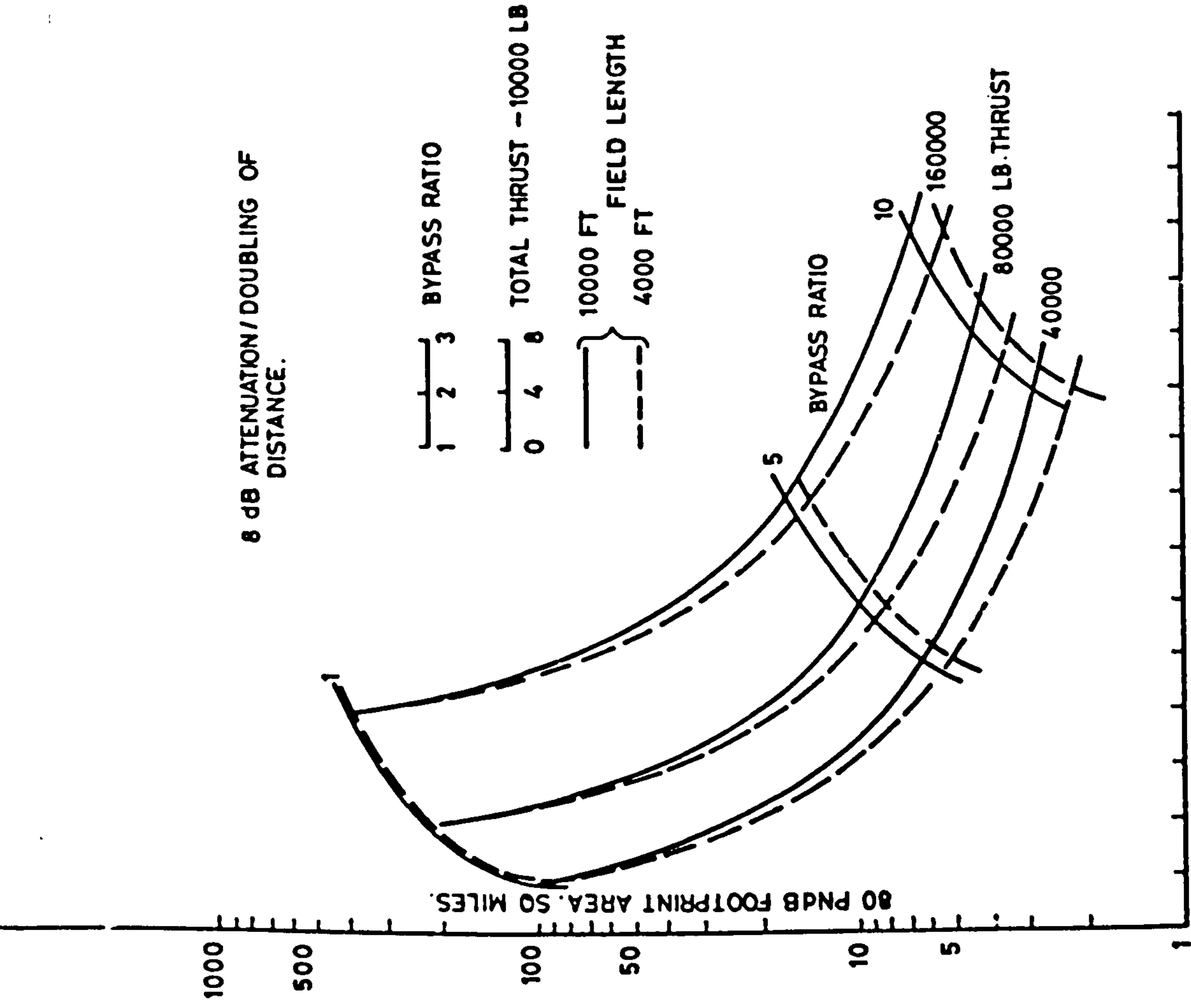


FIG. 7. AS FIGURE 3, WITH 8 dB ATTENUATION.

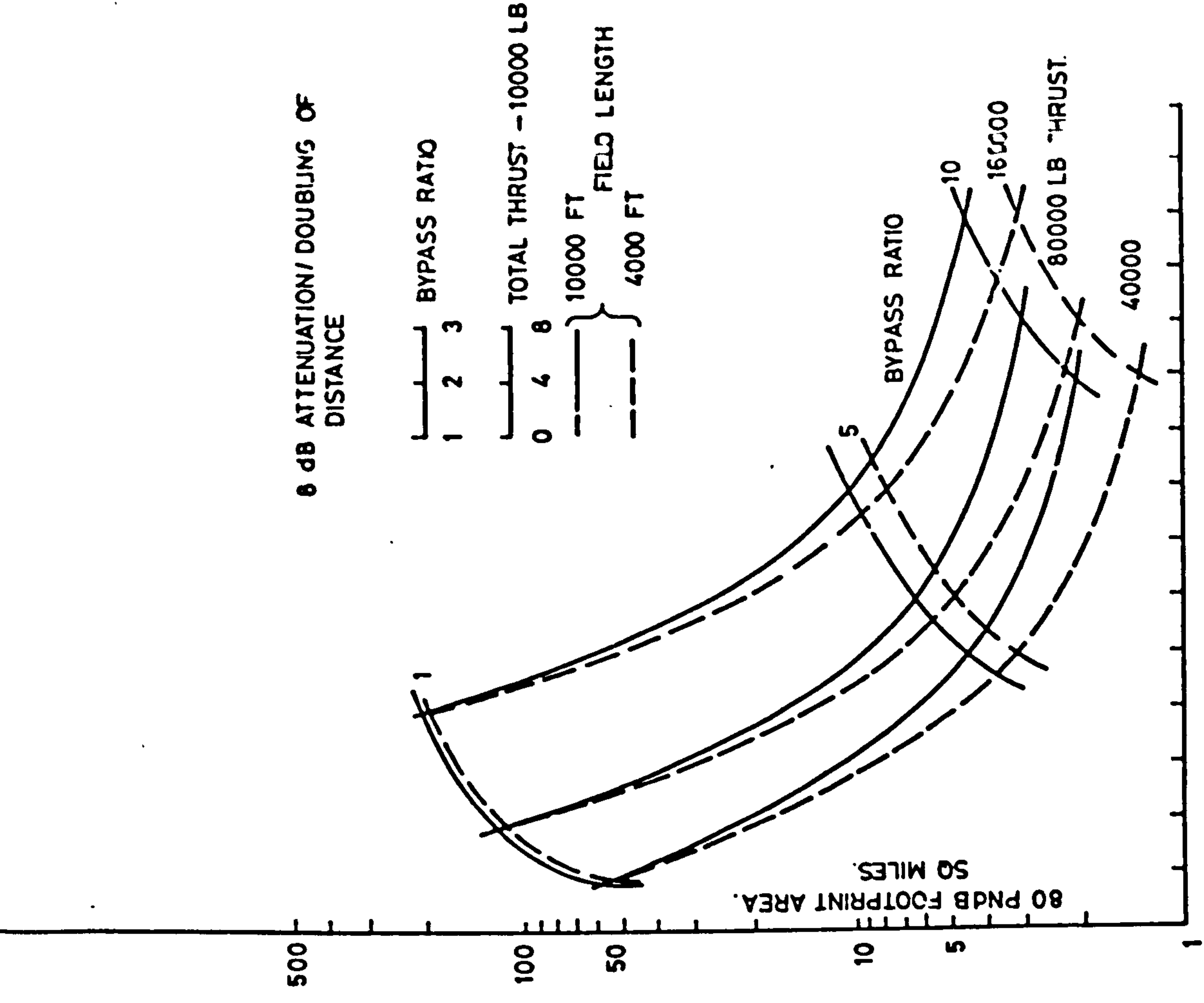


FIG. 8. AS FIGURE 4, WITH 8 dB ATTENUATION.

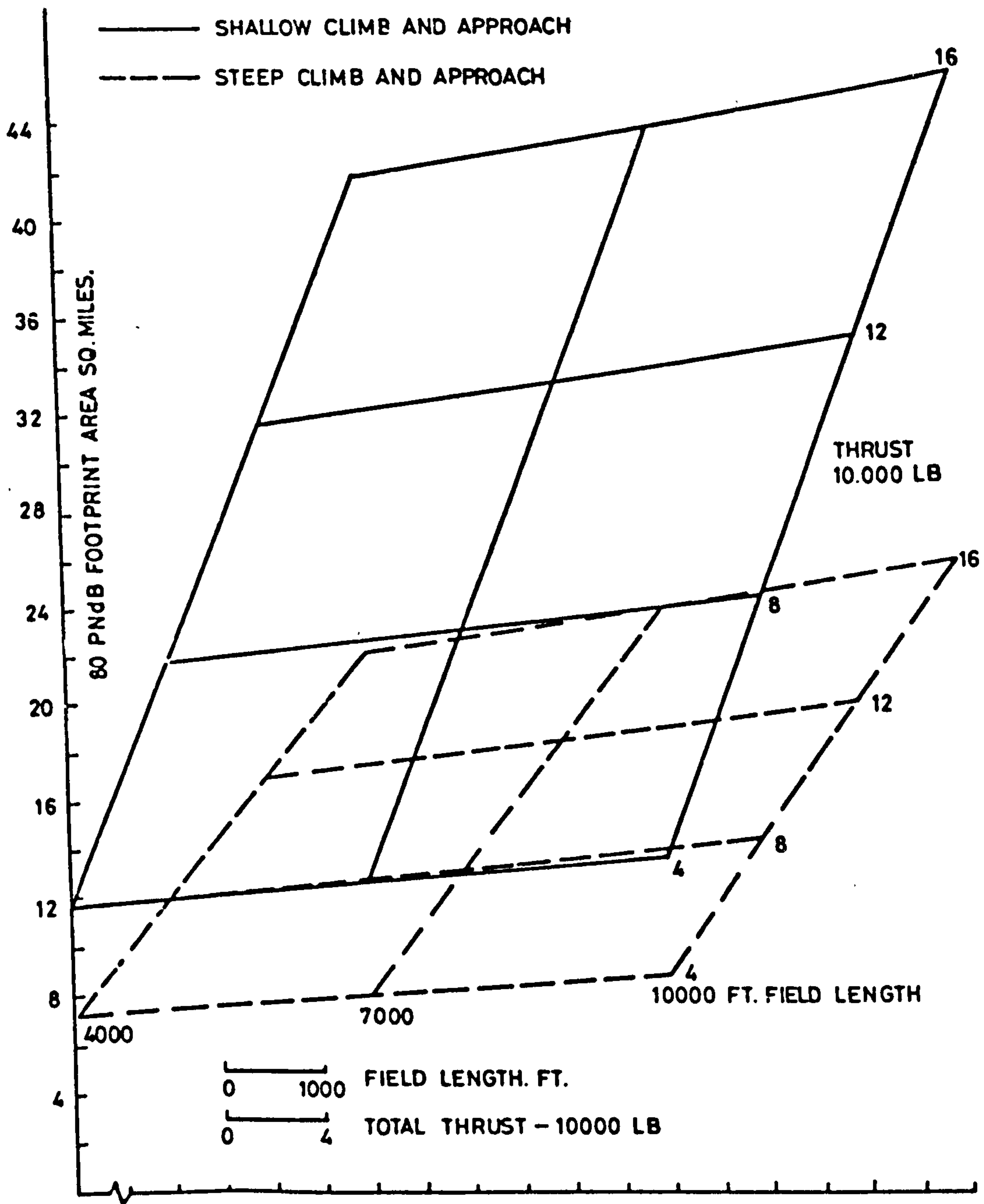


FIG. 9. EFFECT OF FIELD LENGTH ON FOOTPRINT AREA
(STANDARD POWERPLANTS - BYPASS RATIO 5)

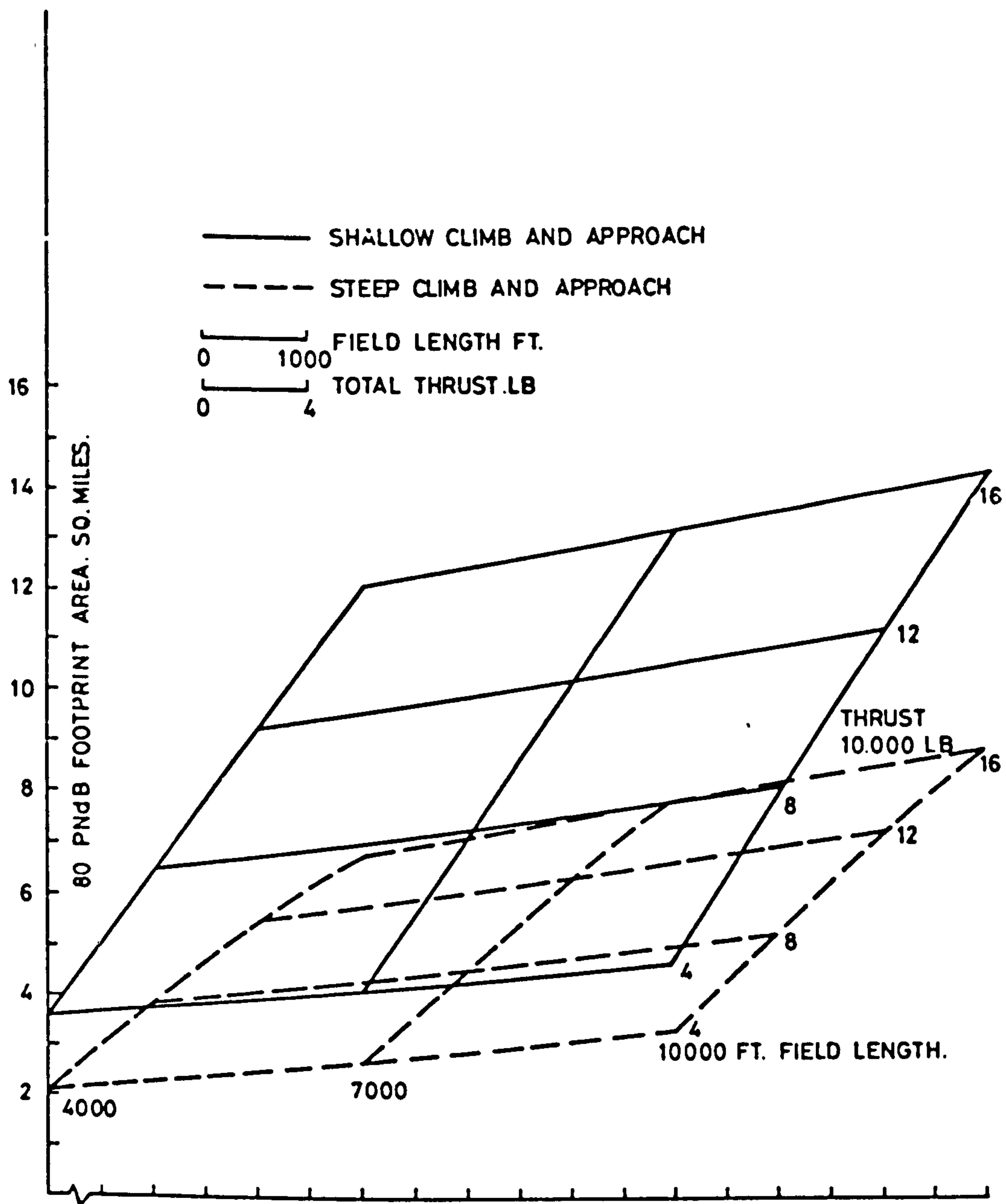


FIG. 10. AS FIGURE 9, WITH QUIETENED POWERPLANTS AND BYPASS RATIO 10.

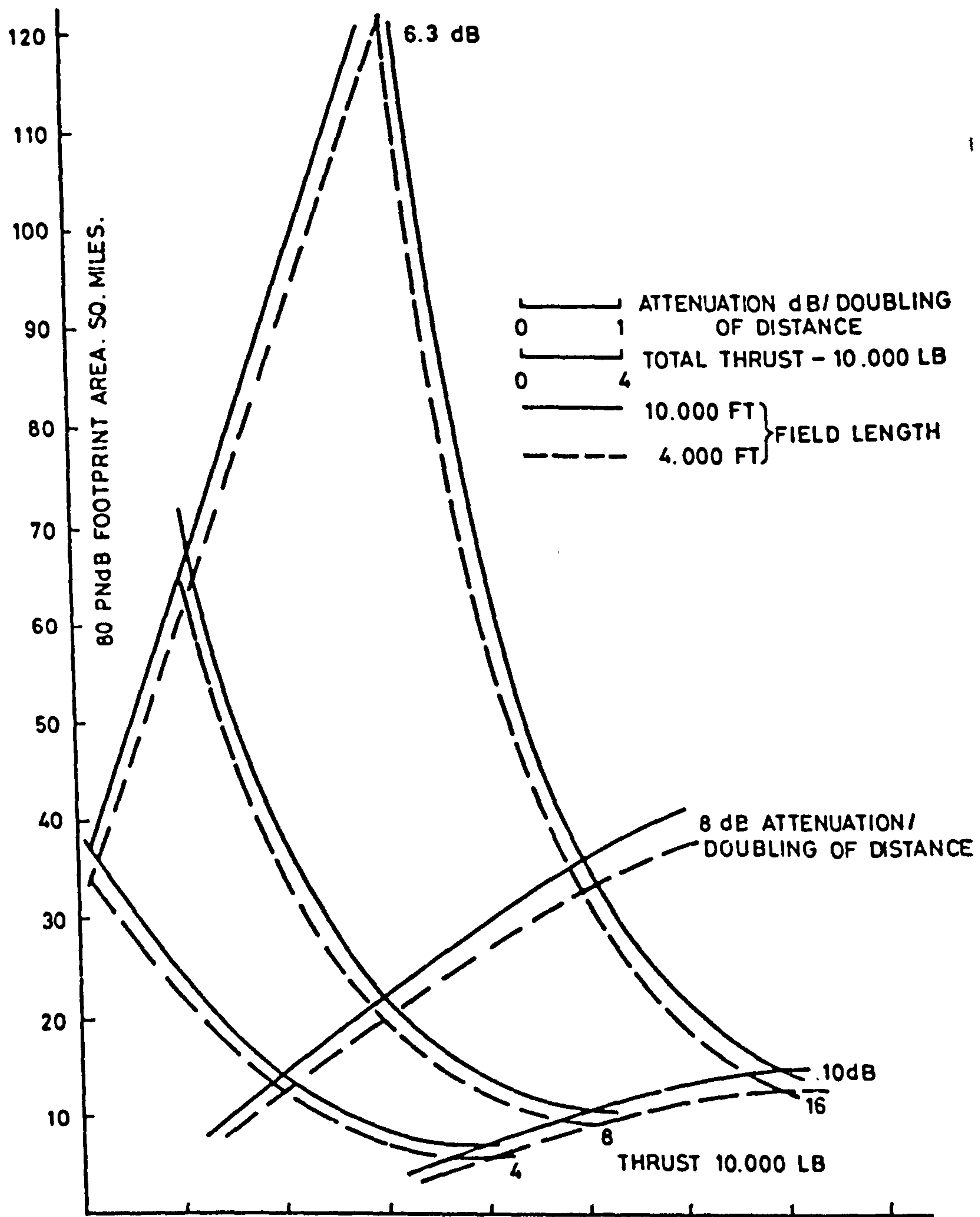


FIG. 11. EFFECT OF ATTENUATION ON FOOTPRINT AREA (STANDARD POWERPLANT - BYPASS RATIO 5 AND SHALLOW CLIMB AND APPROACH).

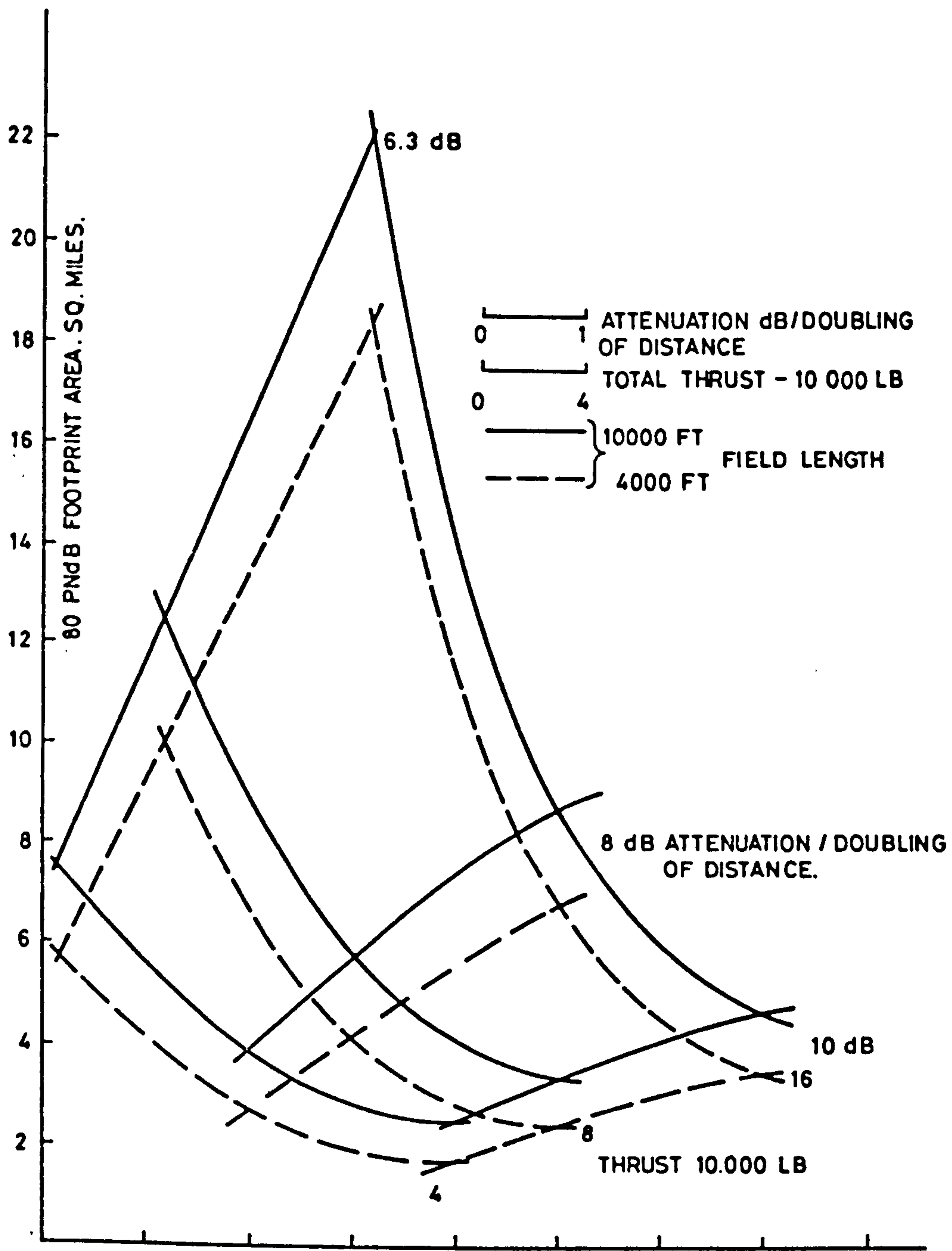


FIG. 12. AS FIGURE 11, WITH BYPASS RATIO 10 AND STEEP CLIMB AND APPROACH.

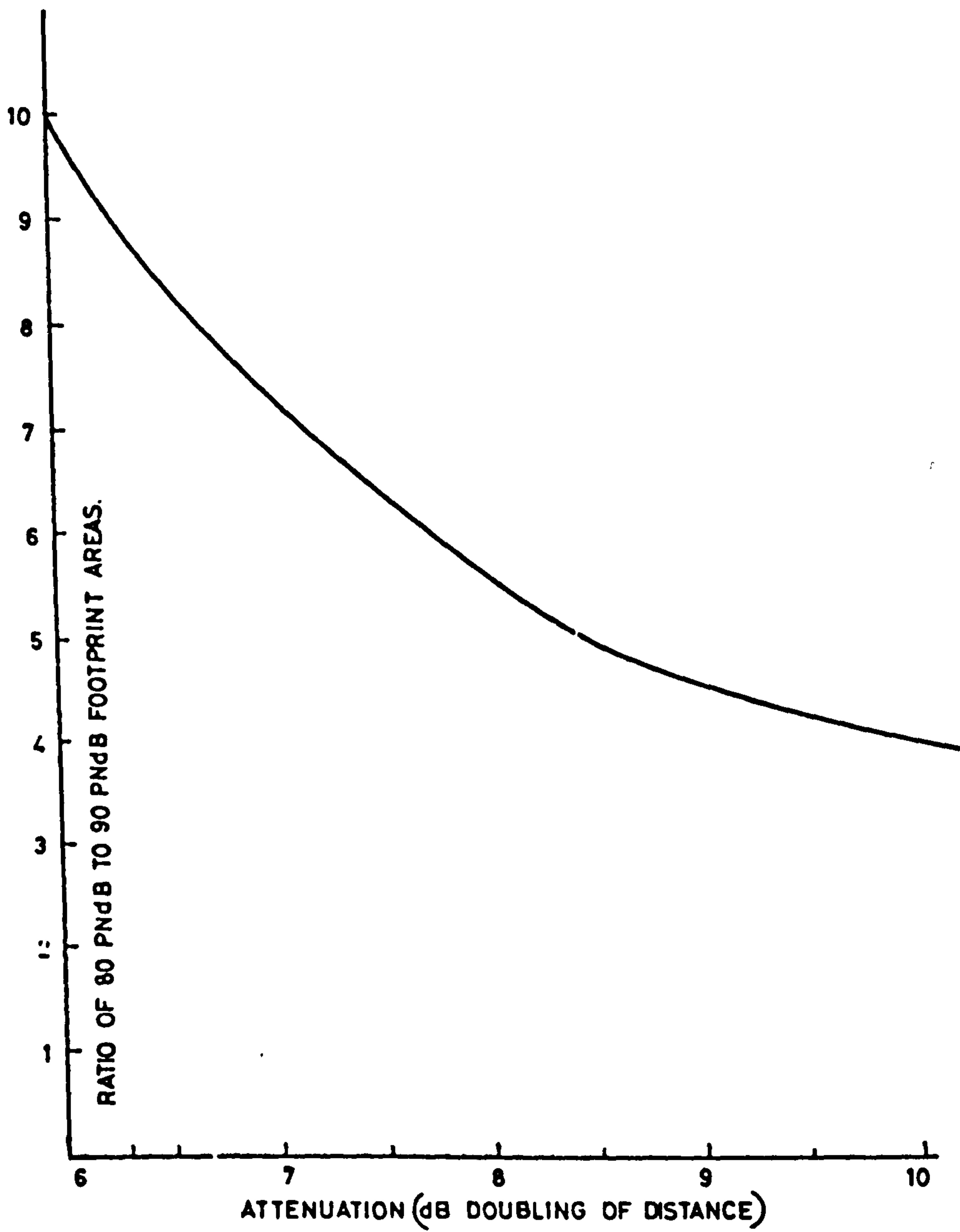


FIG. 13 EFFECT OF ATTENUATION ON RATIO OF 80 PNdB AND 90 PNdB FOOTPRINT AREAS.

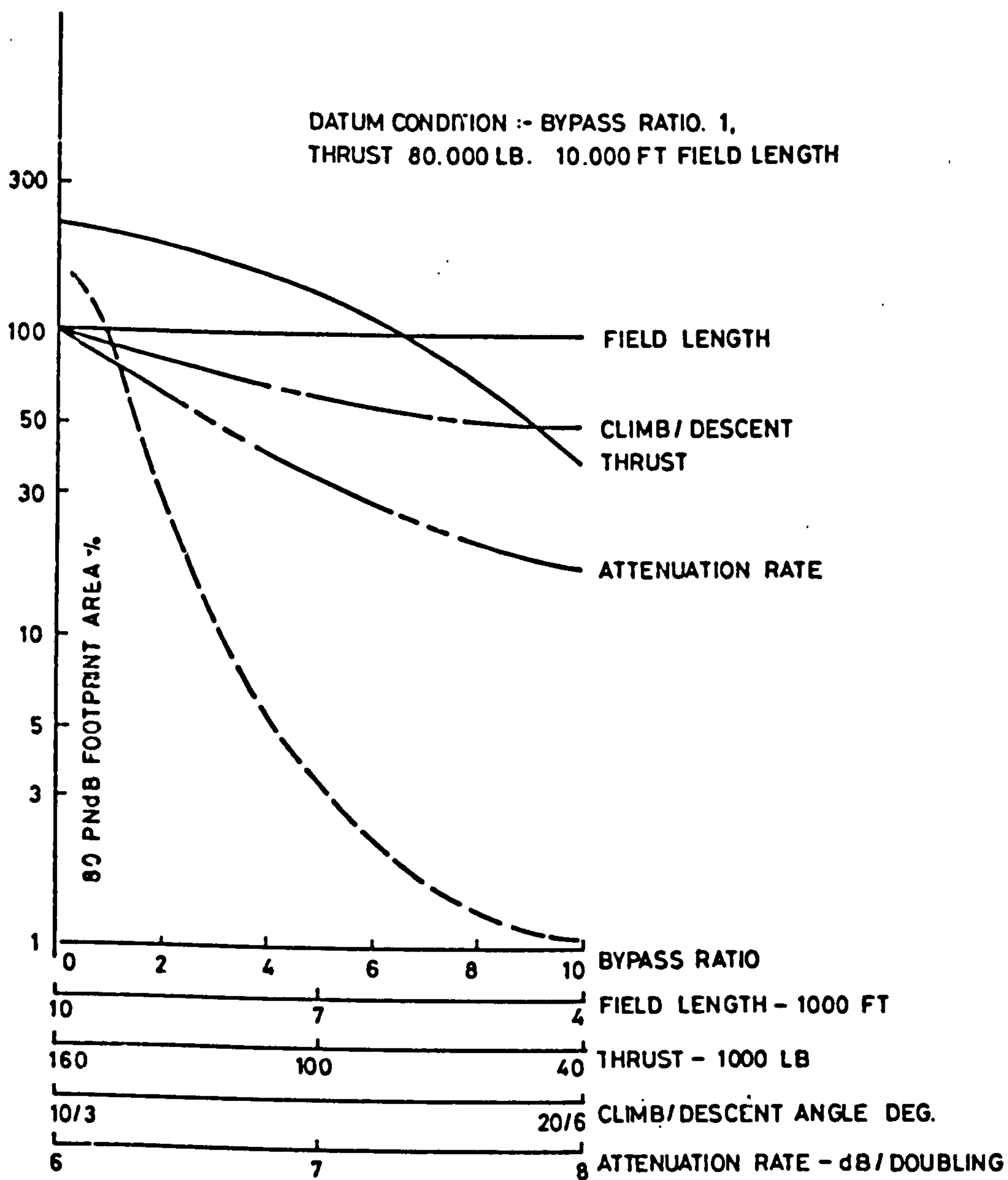


FIG. 14. EFFECT OF PARAMETERS IN REDUCING FOOTPRINT AREA
RELATIVE TO FIRST GENERATION LONG RANGE TRANSPORT
AIRCRAFT CHARACTERISTICS.

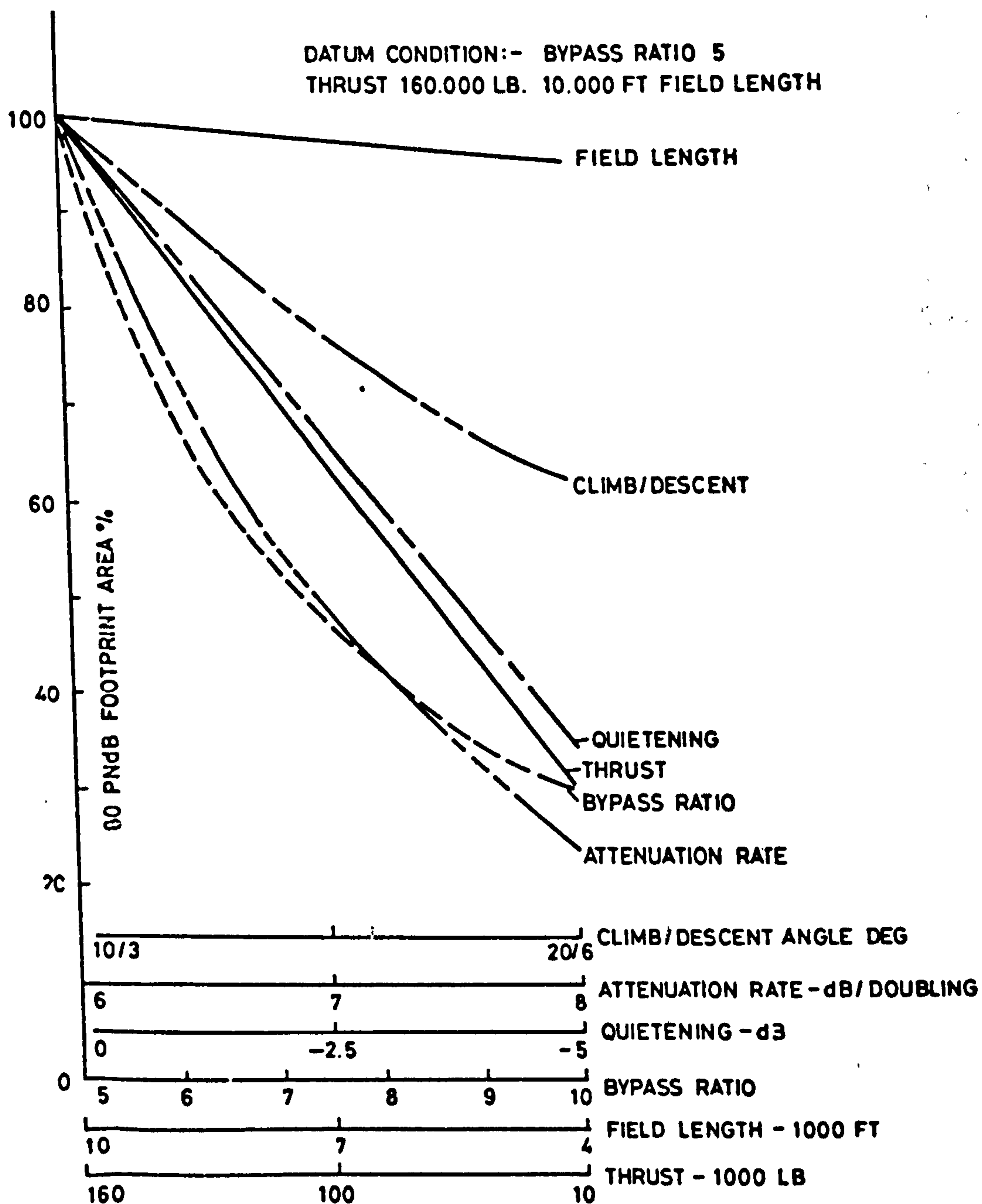


FIG. 15. EFFECT OF PARAMETERS IN REDUCING FOOTPRINT AREA RELATIVE TO WIDE BODY JET TRANSPORT AIRCRAFT CHARACTERISTICS.

January 1974

CRANFIELD INSTITUTE OF TECHNOLOGY

COLLEGE OF AERONAUTICS

THE WEIGHT, ECONOMIC AND NOISE PENALTIES
OF SHORT HAUL TRANSPORT AIRCRAFT
RESULTING FROM THE REDUCTION OF BALANCED FIELD LENGTH

by D. HOWE

SUMMARY

The results of a series of design studies of short haul transport aircraft in the RTOL, STOL and VTOL categories have been analysed to establish their respective performance penalties relative to CTOL types. The main criteria used for comparison are weight, direct operating costs and 80 PNdB noise footprint areas but some consideration is also given to low speed control characteristics. The basis of all the designs was a requirement to carry 108 passengers over a stage length of 600 n. miles plus reserves.

The main conclusions reached are threefold:-

- a) The 4000 ft RTOL design represents an optimum solution if noise is considered to be a prime requirement, in spite of its having significant weight and cost penalties relative to a 5000 ft CTOL design.
- b) The choice for 2000 ft operation lies between the augmentor wing and fan lift STOL concepts.
- c) The tilt wing rotorcraft concept compares well with the fan lift VTOL when high fuel costs are assumed.

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TABLE 1 Summary of Aircraft Characteristics

TABLE 2 Summary of Aircraft Weight Breakdown

FIGURES

NOTATION

For the notation used in the tables and figures see the Key to the tables and Figure 5.

The notation used in the Appendices is as follows:-

b	Wing span (ft)
B	Fuselage breadth (ft)
C_1, C_2	Coefficients used in weight equations
CL_t	Rotor tip lift coefficient
H	Fuselage depth (ft)
L	Fuselage length (ft)
N	Ultimate acceleration factor
R	Engine bypass ratio
S	Wing area (sq ft)
S_b	Rotor blade plan area (sq ft)
SHP	Shaft horsepower
T	Engine thrust, static (lb)
T_R	Reduced engine thrust (lb)
V	Velocity (ft/sec)
V_D	Design diving speed (knots)
V_t	Rotor tip speed (ft/sec)
W	All up weight (lb)
W_a	Weight of empty equipped airframe less engines (lb)
W_F	Fuselage weight (lb)
W_W	Wing weight (lb)
λ	Wing taper ratio
ϕ	Wing quarter chord sweep
τ	Wing root thickness/chord ratio

1. INTRODUCTION

A programme has been undertaken in the Aircraft Design Division at Cranfield during the past four years to study the characteristics of short haul transport aircraft. Initially the emphasis was placed on vertical and short take off aircraft but subsequently the work was extended to include designs for longer runway operation. Although work on these classes of aircraft had been undertaken previously, as for example Ref.1, the present investigation represents a systematic approach to the problem. A general appraisal of the important performance characteristics for this class of aircraft has already been reported (Ref. 2). Particular emphasis was placed on the influence of passenger comfort considerations on runway performance in as much as it affects minimum wing loading and maximum tolerable decelerations. A general conclusion reached was that it should be possible to design aircraft to operate safely from 4000 ft long runways without the need for power augmented lift, and that this should be regarded as the lower limit of reduced take off and landing (RTOL) design. It was shown that the true short take off and landing (STOL) designs capable of operating from runways of the order of 2000 ft length require a substantial degree of powered lift together with a high installed thrust/weight ratio. The possibility of serious low speed control difficulties exists with this class of aircraft due to the inevitably low take off safety and approach speeds although these can be largely eliminated by essentially vertical take off and landing (VTOL) operation. As well as passenger comfort considerations noise and economics were identified as two design requirements of vital importance. However the general comparison of these last two considerations is not readily achieved due to their dependence upon relatively detailed considerations.

The present work is concerned with a comparison of a number of particular designs for short haul transports. These were undertaken as a series of case studies to enable the general conclusions of Ref.2 to be checked and to provide specific information on economic and noise characteristics. An attempt is made to draw general conclusions from this specific information. The studies have all been based on the requirement to carry 108 to 120 passengers over a design stage length of 600 n. miles with a reserve fuel allowance. Cruise speeds varied somewhat according to the actual type of aircraft and the particular operation within the range of 0.67M to 0.83M. Certain of the individual designs have been used as a basis for the annual students' project studies and have therefore been examined in considerable depth. Others have been an individual student's research investigation in which case the emphasis has been placed on their special features. Data derived from existing conventional take off and landing (CTOL) aircraft is used for comparison.

The concepts investigated include fan lift and tilt wing VTOL; fan lift, externally blown flap and augmentor wing STOL; AND RTOL designs.

2. BRIEF DESCRIPTION OF AIRCRAFT

A summary of the more important characteristics of the various designs is given in Table 1. In each case the design weight and thrust has been normalised to a datum of a 108 passenger payload carried over 600 n. miles design range so that direct comparison can be made. The scaling factors used for this process are discussed in section 3. The approach angle was limited to that giving 1000 ft/min rate of descent in each case.

2.1 Fan Lift VTOL (FL)

The fan lift VTOL aircraft used is the A70 design study. The details of this project have been reported fully in Ref. 3 and a general arrangement drawing is shown in Figure 1. In general concept and performance the design has many similarities with the Hawker-Siddeley HS 141 design. The layout is however different in several respects, the reasons for the one chosen being discussed in Ref.3. The two propulsion engines are based on half scale versions of the Rolls-Royce RB211-22 and the twelve lift engines are of the RB 202 family. Although the maximum limited cruise speed is 0.83M the normal condition is 0.78M at 20,000 ft altitude. The assumed fuel reserve allowance includes sufficient fuel for a 100 n. mile diversion and vertical landing after a baulked approach to just above ground level.

2.2 Tilt wing VTOL (TW)

A small tilt wing transport aircraft, known as the E67, was the basis of the annual design project as long ago as 1967 (Ref.4). Whilst this work is not directly relevant to the present investigation the results obtained did suggest that the concept was worthy of consideration for the 108 passenger short haul transport especially as it has potentially a low fuel usage and noise level. An initial investigation was undertaken by Martin (Ref.5) and this confirmed the anticipated advantages. The study is now being taken to a greater depth as the R73 design project with the particular intention of ascertaining the extent of the mechanical complexities of the design (Ref.6). A general arrangement drawing is shown in Figure 2, with the wing in the take off and landing mode. The layout is very similar to the twin rotor convertiplane concepts proposed by Hafner (Ref. 7) and the designs proposed by Westland Helicopters Ltd. Hover disc loading is 36 lb/ft² and each rotor is driven by two 9000 HP turboshaft engines of the Rolls Royce M57HH type. Cruise Mach number is limited to 0.72 but the usual speed at 20,000 ft altitude is somewhat less than this.

2.3 Externally blown flap STOL (BF)

The A71 design study is used as the basis for the externally blown flap versions of the STOL aircraft considered here. The basic A71 work is reported in Refs. 8 and 9, and a subsidiary investigation of the low speed control problems was undertaken by Ward (Ref.10). The layout of the design is shown in Figure 3 and apart from one aspect is very similar to projects proposed for this class of STOL aircraft both in Europe and North America. The flaps are of the double slotted variety. The unusual feature is the wing mounted nacelle used for housing the long stroke undercarriage. As originally designed the A71 was powered by 4 engines of the Rolls Royce M 45S (RB410) type to give it a static thrust ratio of 0.5. The engines have a variable pitch fan with a bypass ratio of rather more than 10. The intention was that the aircraft should be capable of operation from runways of 2000 ft length but the detail performance evaluation revealed that the engine failure case on take off precluded this due to the consequent loss of both lift and thrust. Increase of lift off speed to give the required margin of safety necessitated the take off balanced field length being increased to approximately 2600 ft although there is adequate margin for landing on to a 2000 ft runway. Safe take off from a 2000 ft runway requires the static thrust/weight ratio to be increased to nearly 0.7 with a consequent overall increase in aircraft size and weight. It is this modified version of the A71 which is used in this report as the datum 2000 ft externally blown flap STOL design. The Mach limited cruise at 30000 ft altitude is 0.8 but a more useful condition is rather less than 0.7M at 20,000 ft altitude. This lower cruise speed is determined by cruise comfort conditions resulting from the relatively low take off wing loading of 74 lb/sq ft.

2.4 Augmentor wing STOL (AW and QAW)

A preliminary study of the effect of replacing the externally blown flaps of the A71 by the augmentor wing concept was made by Van Twisk (Ref. 11). For simplicity the basic A71 design was used as the datum and the powerplants replaced by units of the RB 419 type with the addition of a facility for tapping large quantities of air for the lift augmentation system. The additional gas generator size necessary to enable this to be done results in a thrust increase of about 30%. This extra thrust and the relatively higher efficiency of the augmentor wing resulted in an aircraft capable of operation from 1600 ft runways. The nozzle augmentor pressure ratio is approximately 1.9. At the same time the low speed control problems, especially those associated with the engine failure case, are likely to be less severe. The penalty for these advantages lies in the greater all up weight and the mechanical complexity of the flap system. The degree of the latter and the low speed control characteristics are the subjects of a current study. Landing rather than take off was found to be critical in determining the runway length.

A further potential advantage of the augmentor wing relative to the externally blown flap is the possibility of reducing noise levels by the use of multi-element nozzles

and acoustic lining within the flap segments. This feature has been investigated by the Boeing Company as part of an STOL project study (Ref.12), use has been made of this information to derive the quiet augmentor wing (QAW) as a somewhat larger and heavier aircraft.

2.5 RTOL (UW and RE)

The basic A71 design has also been used as the starting point of the design investigation for an RTOL aircraft but in this case there are numerous differences in the final layout. This work was carried out by Jesse (Ref.13). The major differences consist of replacing the double slotted flaps of the A71 by triple slotted units and drooped ailerons, increase of the wing loading to 84 lb/sq ft and reduction of the static thrust/weight ratio to 0.46. The latter requirement may be met either by two or four underwing engines (UW) in which case the layout is similar to the A71 or by two or three rear fuselage mounted engines (RE). The merit of the rear engine version is that it enables a low wing to be employed with consequent advantages in the flap and undercarriage layout. It also introduces the possibility of using the wing to assist in noise reduction by employing it to shield the intakes. The three engined layout proposed in Ref.13 appears in Figure 4. Noise shielding could be carried further by introducing a low mounted tailplane in association with two powerplants. In all cases the powerplants are of the Rolls Royce M53 family. Take off requirements dictate the use of a runway of 4000 ft nominal length with the design landing requirements being somewhat less. The higher wing loading relative to the A71 enables the normal cruise Mach number at 20,000 ft to be increased to 0.73 without a reduction in passenger comfort.

2.6 Fan lift STOL (FL)

One way of achieving RTOL or STOL performance is to use a number of fan lift engines to give a vertical thrust component thereby augmenting the wing aerodynamic lift. The layout and concept of the aircraft is thus an exact intermediate between the conventional and fan lift VTOL designs unlike other STOL types where the lift augmentation is indirect. For purposes of comparison a 2000 ft runway STOL design has been derived by the simple device of adding six fan lift engines to the 4000 ft RTOL layout discussed above. It is visualised that the additional powerplants would be mounted along the sides of the fuselage fore and aft of the structural box of the low mounted wing. During take off the lift engines would be inclined at approximately 30° to the vertical to provide a substantial forward thrust component. The resultant equivalent static thrust/weight ratio is about 0.63. This design is not necessarily an optimum 2000 ft STOL fan lift aircraft but is regarded as a possible development from the 4000 ft RTOL.

2.7 CTOL (UW and RE)

Existing data from aircraft such as the Boeing 737 and BAC 1-11 has been used to derive datum CTOL aircraft of underwing and rear fuselage powerplant layout respectively. It has been assumed that the powerplants used are of the M53, bypass

ratio 10 family and that the normal cruise speed at 20,000 ft is approximately 0.7. A nominal runway length of 5000 ft has been regarded as the datum value for comparative purposes, but the effect of relaxing this to 7000 ft has also been considered. It is worth noting that the Boeing 737/200 can carry the design payload of 108 passengers over the 600 n.mile stage length when operating from 5000 ft long runways. The take off weight is approximately 90000 lbs in this condition.

3. WEIGHT SCALING

Although all the project studies compared here have very similar design requirements certain small differences did exist in the basic specification. It has thus been necessary to undertake a weight scaling process to bring them all to a common base and at the same time to make allowance for different augmentation systems, engine location, etc. This was done by using simple empirical relationships appropriate to short haul transport aircraft to modify the component weights estimated from the detailed design studies. The formulae used were derived from Ref.14 and the assumptions made are stated in Appendix A. No weight scaling was required for the VTOL designs.

The final weight breakdowns are given in Table 2. The gross weights related to that of the datum 5000 ft CTOL aircraft with underwing powerplants are shown as a function of balanced field length, in Figure 5.

Only in the case of the augmentor wing was allowance made for the weight of noise reduction techniques. In other cases it was considered that the necessary development would be achieved within the weights predicted. This is discussed in Appendix A.

4. LOW SPEED OPERATING CHARACTERISTICS

Brief comments on the low speed operating characteristics of the various designs are of interest. The variation of the approach and take off safety speeds with balanced field length are shown in Figure 6. The major limitation imposed upon the augmented lift STOL designs arises from the loss of lift as well as thrust when engine failure occurs. This is more serious during take off than landing since in the latter case it is possible to open the throttles of the remaining functioning engines to compensate for the loss although it is necessary to cope with a missed approach. For balanced field lengths down to just under 2000 ft the take off case is more critical. However for field lengths below 2000 ft the landing case becomes more important in establishing the required length of runway due to the comfort limit imposed on the mean design braking deceleration (Ref.2). The augmentor wing design requires a 1600 ft long runway for landing rather than take off reasons.

Two particular problems associated with reduced take off and approach speeds are the control problems which arise when engine failure occurs or when there is a large cross wind component. The severity of the engine failure case depends upon the design layout and nature of any lift augmentation system

employed. Inevitably it is severe for the externally blown flap concept (Ref.10) where it gives rise to very high roll and yaw control demands. The use of an augmentor-wing reduces the magnitude of this problem by virtue of the possibility of using cross ducting for the blowing air to tend to equalise both the lift and thrust distributions. Van Twisk (Ref.11) has shown that the potential improvement is considerable and this is why the more detailed analysis of this and other low speed characteristics of the augmentor wing concept is being undertaken. In the case of the fan lift designs it is reasonable to assume that the difficulty is overcome by ensuring that the number of lift engines used is adequate. For the vertical take off case the minimum number appears to be twelve. The four engines of the tilt wing design are mechanically interconnected and adequate emergency power is available to cope with the case of single engine failure during take off. A measure of the severity of the cross wind problem can be gauged by reference to Figure 7, which shows the cross wind component as a function of balanced field length corresponding to two particular equivalent yaw angles on the approach. The lower of these, 12.5 degrees is representative of current practice whilst the higher, 20 degrees probably represents the absolute limit and may well imply some form of castoring main undercarriage. It is shown in Ref.2 that in the case of aircraft operation from exposed single runway aerodromes the mean cross wind is likely to exceed about 22 knots for a significant number of hours per year. Thus an unusually severe problem exists for aircraft designed to operate from balanced field lengths of less than 4000 ft. From the control point of view it is necessary to be able to deal with the lateral gusting associated with the cross wind condition. Ref. 10 indicates that the gust velocity is just under half the mean cross wind component.

5. DIRECT OPERATING COST EVALUATION

A comparison of the direct operating costs of the different designs has been undertaken. The B.E.A. method (Ref.15) was used as a basis for this but it was necessary to make a considerable number of changes to cater for the different types of design and probable escalation of costs. All the assumptions made are stated in Appendix B and the results are summarised in Table 1. These have been based on the case of a one hour block time and a fleet of 20 aircraft of any given design. The one hour block time implies a sector length of 300 n. miles in all cases except that of the lift fan VTOL where the higher cruise Mach number results in a sector length of 360 n.miles. Two different fuel costs have been used in order to establish the sensitivity of the direct operating costs to this parameter. The lower of these referred to as fuel cost A is 1.5 p per lb and the higher, fuel costs B, is 4.5 p per lb. Whilst it is impossible to forecast fuel costs with any degree of certainty at the present time it is hoped that these values relative to the other costs do cover the range likely to be experienced within the time scale appropriate to the study, that is in the decade beginning about 1980. The relative fuel loads are shown in Figure 8, both on a total provision and sector basis.

The direct operating costs of the fan lift designs are initially dependent upon the prime costs of the engines and the spares carried and it is considered that reasonably optimistic assumptions have been made for these values. Likewise the costs of the tilt wing design depend very much upon data associated with transmission and rotor systems. The assumptions made here were based on rather sparse evidence and are therefore open to criticism. It is hoped that more accurate information can be estimated when the detail design work on the concept has been completed but in the meanwhile it is felt that the results obtained form a reasonable basis for comparison with other designs. In evaluating the engineering costs an attempt was made to allow for such items as the complexities of flap and control systems and changes in undercarriage operation. It was found that with the possible exception of the fan lift VTOL design the net changes were of negligible significance and even in the exception the overall effect was well within the anticipated accuracy of the total calculation.

Figure 9 shows the variation of direct operation costs with balanced field length and also the variation of the ratio of the costs to gross weight. The different fuel costs have negligible relative effect except for the VTOL designs.

6. NOISE CHARACTERISTICS

The noise characteristics of the design have been compared using the best available, consistent, information. The assumptions made and the source of references used are given in Appendix C. The comparison is based upon the noise level at 500 ft on the sideline and where appropriate below the flight path, and more particularly on the area of the 80 PNdB noise footprint. The latter is regarded as the real criterion for aircraft operation from urban located sites in the future; see for example Ref. 23. A very important assumption in the evaluation of the footprint area is the rate of sound attenuation. The 500 ft noise levels were based on published data much, if not all, of which assumed a 6.3 dB attenuation for each doubling of the distance from the source. However there is reason to believe that the ground level attenuation at least is greater than this. Therefore it was assumed that the attenuation beyond 500 ft is at the rate of 8 dB for each doubling of the distance, see for example Ref. 22. In the noise evaluation an attempt was made to allow for the results of engine and airframe developments in the direction of noise reduction. These are discussed in Appendix C and in most cases have resulted in two sets of noise figures appropriate to 'existing' and 'quietened' designs. It is necessary to note that even the 'existing' design assumptions do anticipate significant improvements relative to current operational aircraft. The comparisons are summarised in Table 1 and shown in Figures 10 and 11. The latter of these shows the relative 80 PNdB footprints for the quietened designs whilst the former gives the absolute values of the areas for both cases.

The transition altitude for all the VTOL examples was taken to be 2400 ft, with the flight up to and down from this condition being essentially vertical. The elongation of the circular footprint due to the climb away was, of course, allowed for. The airframe noise was found to be significant in the approach noise level of the noise shielded designs, that is those cases where engines are arranged relative to the lifting surfaces to give a blanking effect. In these cases the perceived airframe and engine noise components on the 500 ft sideline are approximately equal. A reduction of 14dB was assumed for the quietening of the augmentor wing flap system and this brought the noise from this source to below that of the basic powerplants.

7. DISCUSSION

7.1 General

The relative smoothness of the variations of the weight, costs and noise with field length is encouraging. There are, of course, points which are off the curves and these are the result of a difference in concept, as with the tilt wing, or indicate a particular characteristic of note. The trends of the curves are as anticipated.

7.2 Relative weights (Figure 5)

The designs studied show clearly that increase of gross weight associated with reduction in field length. Relative to the 5000 ft CTOL with underwing powerplants there is about 12% weight penalty for each 1000 ft reduction in field length down to 2000 ft. It is interesting that below this distance the penalty is proportionally less severe which may well be due to the use of more directly derived powered lift. For example considering the fan lift engine designs relative to the 4000 ft RTOL with rear fuselage powerplants the weight penalty is only about 7% for each 1000 ft reduction in field length. The rear engine noise shielded designs are about 3% heavier than the underwing powerplant aircraft and the 4000 ft RTOL referred to above is some 16% heavier than the datum.

Of the two VTOL concepts the tilt wing is some 5% lighter than the fan lift engine version. This is due to the use of the one set of powerplants for both vertical and forward flight and the lower fuel requirements. The heaviest aircraft is the quiet augmentor wing. This is some 4% heavier than the basic version and about 1% heavier than the fan lift VTOL though it requires a 1600 ft field length.

7.3 Low speed operating characteristics (Figures 6 and 7)

It is clear that aircraft designed to operate from field lengths of less than 4000 ft suffer from increasingly more severe low speed problems unless they are designed for essentially vertical operation. Of the STOL designs the augmentor wing introduces a smaller engine failure problem than the externally blown flap design but the fan lift concept could be even better due to the greater scope available in layout. The approach crosswind problem is a function of the nature of the operation as well as field length and is not primarily dependent upon the particular design concept.

7.4 Fuel requirements (Figure 8)

With the probable long term restriction and high cost of fuel supplies the fuel requirements become a particularly important consideration. The 4000 ft RTOL requires about 10% more fuel than the 5000 ft CTOL aircraft used as the datum. However reduction in field length below 4000 ft necessitates the provision of about 20% extra fuel for each 1000 ft reduction in runway length as far as the basic family of aircraft is concerned. The fuel actually used during the one hour flight assumed for cost evaluation is relatively greater for the fan lift designs. However it must be remembered that the VTOL version actually flies 12% further on the fuel used, so that on an aircraft-mile basis it requires 75% more fuel than the datum design. The tilt wing concept is relatively efficient from this point of view and on the evidence available uses only about 6% more fuel than the 5000 ft CTOL. This is presumably due to the very much higher effective bypass ratio of the powerplant/rotor system.

7.5 Direct operating costs (Figure 9)

The trend of the direct operating costs follows closely the trend of take off weight although there is an indication of a somewhat greater relative penalty for the shorter runway design. The two exceptions are the fan lift STOL and the tilt wing VTOL. The former has a consistent operating cost relative to other 2000 ft concepts in spite of its lower weight. The assumptions made for the latter indicate that it is relatively expensive to operate and as has been noted previously there may be an undue weighting against this design due to lack of precise data. However it is important to note that if fuel costs rise considerably it could be less expensive to operate than the fan lift VTOL, even on the basis of the assumptions made.

The basic curve suggests a 14.5% penalty on direct operating costs for each 1000 ft reduction in balanced field length in the range of 5000 ft down to 2000 ft, and about 60% penalty for VTOL relative to the datum if the higher fuel costs are assumed.

7.6 Noise (Figures 10 and 11)

The 80 PNdB noise footprint areas shown in Figure 10 have been included to show the absolute values predicted for this important parameter, for the cases of both shorter and longer term development. By and large the figure show considerable improvement relative to existing aircraft. For the purposes of the present investigation the relative 80 PNdB footprint areas shown in Figure 11 are of more significance. The data in this case applies to the longer term developments as this is regarded as being more justified for the newer design concepts.

The first point of note is the very unfavourable characteristics of the externally blown flap design. It is very difficult to visualise any means of improving this and it does seem that this concept must have a footprint area which is some five times that of the other designs examined.

There is relatively little variation in the other cases. Unlike the curves shown for weight and cost there is some evidence of an optimum field length. It is not a particularly strong tendency but occurs at about 4000 ft RTOL. The noise shielded designs do show significant improvements relative to the comparable underwing ones. This amounts to about 25% less area for the 5000 ft design and over 30% at 4000 ft. The tilt wing VTOL footprint area is rather less than 70% of that of the fan lift VTOL.

Unlike the externally blown flap design the augmentor wing in its quiet version has very similar characteristics to the fan lift aircraft and like them is not very much worse than the datum design.

7.7 Overall comparisons

In an attempt to compare the designs on a more comprehensive basis two merit indices have been introduced. The first of these is essentially an environmental one since it is the product of the relative fuel requirements and the 80 PNdB footprint ratio. The variation is shown in Figure 12. As would be expected the externally blown flap and basic augmentor wing have undesirably high values of the index. The underwing engine family has the datum index of unity for balanced field lengths above about 4200 ft and below this the index increases by about 0.3 for each 1000 ft reduction. The rear engined 4000 ft RTOL is the best with an index of 0.75 but the tilt wing VTOL compares very favourably with an index of 0.85.

The second index is based on the direct operating costs and is therefore classified as an economic merit index. The 80 PNdB footprint ratio is included again since this must be of prime importance in any future design. The values of this index are given in Figure 13 where it can be seen that the general pattern is very similar to the previous one. The main differences are at the VTOL end of the spectrum where the tilt wing shows up less favourably than before but is still better than the fan lift design. There is relatively little between the fan lift and quiet augmentor wing STOL and the VTOL concepts.

In general it may be concluded from these two figures that the 4000 ft RTOL has much in its favour, particularly in the noise shielded rear engine version. It should not introduce any severe low speed control problems. In the light of the present work it represents the best compromise between the various environmental and economic considerations and in this sense may be regarded as an optimum.

Should STOL applications at around 2000 ft field length be required the choice lies between the quiet augmentor wing and fan lift concepts. Both have development problems associated with them, although the augmentor wing may prove to be the less severe of the two since it can be approached more gradually than the production of a complete new lift engine design. On the other hand the fan lift STOL is a natural step on the road to VTOL which cannot be said for the augmentor wing.

Perhaps one of the more interesting results is the potential shown by the tilt wing. In spite of its undoubted complexity and the consequential high operating costs it does have favourable

fuel and noise characteristics. In the present climate these are likely to prove the dominant issues. At the very least the tilt wing concept deserves a renewed appraisal and the same comment could be made in the context of other rotorcraft designs, such as the blown rotor. However having said this it must be admitted that the fan lift VTOL does have a greater speed potential and hence work capacity. It can be derived more directly from current transport aircraft practice once the lift engine is available. In its developed form the noise footprint should be acceptable and the main disadvantage is the high fuel consumption.

8. CONCLUSIONS

8.1 Although the 4000 ft RTOL concept implies some increase in direct operating cost relative to more conventional designs, when consideration is given to noise characteristics it represents an overall optimum. This is especially true for a rear engine, noise shielded layout.

8.2 The externally blown flap concept is ruled out when noise is of any importance. Its low speed control problems are also more severe than those of the augmentor wing.

8.3 For STOL operations from fields of about 2000 ft length there is little to choose between a quietened augmentor wing and a fan lift design. The former may prove to be a more straightforward development but the latter is a natural step towards VTOL.

8.4 The tilt wing concept shows up very promisingly when the basis of comparison is environmental, that is fuel usage and noise. The results suggest that this concept and other forms of rotorcraft should be reviewed in the present, changed circumstances. However in spite of its high fuel consumption the fan lift VTOL should be acceptable from a noise point of view and has a greater work capacity than rotorcraft.

8.5 The 80 PNdB noise footprint areas of the CTOL aircraft have been estimated to be similar to those of the RTOL, STOL and VTOL designs on the basis of developments in the noise reduction likely to be achieved in the next decade or so. This is different to earlier predictions and arises primarily from the use of quietened powerplants of high bypass ratio.

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APPENDIX A

WEIGHT SCALING FACTORS (Ref.14)

1. WING WEIGHT

The weight of the wing, including the flap system for a conventional design can be expressed as:-

$$W_W = C_1 \left[\frac{bS}{\cos\phi} \left(\frac{1+2\lambda}{3+3\lambda} \right) \left(\frac{WN}{S} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \text{ lbs}$$

where b is the wing span, S the area, λ the taper ratio, τ the thickness/chord ratio at the root, N the ultimate normal acceleration factor, ϕ the quarter chord sweep and V_D the design diving speed in knots. All dimensions are in feet units and W is the all up weight in lbs.

C_1 is 0.003 approximately when the engines are carried on the wing and 0.00315 when they are located elsewhere. For a consistent wing geometry, wing loading and design requirements this yields:-

$$W_W \propto W^{1.35}$$

This was used in conjunction with the weights derived from the detail investigations but it was also necessary to make a correction to allow for the major differences in the flap systems used. This has been carried out in absolute terms using the following values of weights per unit planform area of the flaps and slats:-

Double slotted trailing edge flaps	4 lb/sq ft
Triple slotted trailing edge flaps	6.5 lb/sq ft
Augmentor flaps	8.0 lb/sq ft
Kruger flaps	5.0 lb/sq ft
Leading edge slats	7.0 lb/sq ft.

2. FUSELAGE WEIGHT

Fuselage weight can be expressed as:-

$$W_F = C_2 \left[L(B+H)V_D^{0.5} \right]^{1.5}$$

where L is the fuselage length, B the breadth and H the height, in feet.

C_2 is 0.001 normally, but 0.0011 when the powerplants are fuselage mounted. In fact in this case it was only necessary to use this as a correction on the established design weights.

3. TAIL UNIT

For the present purposes tail unit weight was assumed to be proportional to $W^{0.83}$.

4. UNDERCARRIAGE

The weight of the undercarriage was taken as being directly proportional to all up weight for consistent geometry

5. SYSTEMS, INSTALLATIONS AND EQUIPMENT

Fuel system and the flying control system were each assumed to be proportional to all up weight. Air conditioning and de-icing were taken together and allowance made for change in wing and tail area. All other items of equipment, installations, disposables, etc. were assumed to be constant.

6. POWERPLANT AND FUEL

Powerplant weight was assumed to be proportional to thrust and hence to all up weight for a given static thrust/weight ratio. The gross installed weight of the bypass ratio 10 powerplants was taken as 0.275 times the static thrust. In the case of fan lift engines the gross installed weight, excluding nacelle structure, was assumed to be 0.076 times the thrust. Lift engine nacelle structure for the 2000 ft fan lift STOL design was deduced to be 2000 lbs from the estimated weight of the fan lift VTOL aircraft, the A70. Similar deductions for pylon and propulsion engine nacelle weight were made from the other design studies.

Fuel weight was assumed to be proportional to the gross weight for the small weight variations associated with the scaling process.

7. WEIGHT PENALTY FOR QUIET DESIGNS

In the case of the propulsion engines it was considered to be reasonable to assume that developments in technology would enable approximately 3.5 dB reduction in noise level without significant weight increase above that already provided.

The improved noise level of the lift fan engines was assumed to be 6 dB less and due to the use of silencers. There is no doubt that these would involve a weight penalty but it was assumed that provision was already made for this in the gross installed weight allowance. Thus the 'existing' design weights may be considered to be somewhat high but in reality are probably not too unrealistic for early production engines.

The internally blown flap design has a very high noise level although it may possibly be reduced by about 5 dB by using mixers on the powerplants. No weight allowance has been made for this since the whole issue is very tentative.

On the other hand specific information is given in Ref.12 on the penalty for noise reduction of an augmentor wing system by using acoustic lining and multielement nozzles. This reference implies a weight penalty of about 0.14% of the all up weight for each 1 dB noise reduction up to 14 dB. This figure has been used in evaluating the data for the Quiet Augmentor Wing STOL.

APPENDIX B

DIRECT OPERATING COSTS (Ref.15)

The direct operating cost evaluation was based on the BEA method modified to cover the different types of aircraft and possible future cost evaluation. The evaluation was based on a fleet on 20 aircraft in each case and 100% load factor.

1. BLOCK TIME, UTILISATION AND SECTOR DISTANCE AND FUEL

The block time was assumed to be 1 hour in each case resulting in an annual utilisation of 2200 hours. With the exception of the fan lift VTOL each aircraft was assumed to cruise at 300 knots EAS at 20,000 ft to give a sector distance of 300 n. miles and hence a block speed of 300 knots true. Analysis of the performance of these designs carried out in detail indicated that for this case the sector fuel was 36.5% of the total provided with the 108 passenger payload. In the case of the fan lift VTOL the cruise speed is higher, being 350 knots EAS and in this case the sector distance is 360 n.miles with a block speed of 360 knots true. The fuel used was found to be 42% of the total provided.

2. PRIME COSTS

The prime cost of the equipped airframe was taken to be basically £50 per lb of As Prepared for Service weight. However because of the very high powerplant content of some of the designs a correction was applied to allow for this on the basis of the assumed powerplant costs. This correction was found to have little effect apart from the case of the fan lift aircraft. The 2000 ft fan lift STOL was corrected to £51.8 per lb and the fan lift VTOL to £56.8 per lb.

Propulsion engines were assumed to cost £15 per lb of static thrust and fan lift engines £7 per lb of static thrust. This last figure is very critical in determining the operating costs of the fan lift aircraft.

The 9000 HP shaft engines used in the tilt wing VTOL design were each assumed to cost £140,000, less gearbox. The gearbox unit couples pairs of engines and together with the cross shafting was estimated at £160,000 each. Rotor unit costs are somewhat problematical to predict but were assumed to amount to £320,000 each.

3. SPARES

Airframe spares investment was taken as 12% of the prime cost of the equipped airframe.

Engine spares holdings were assumed to be as follows

Propulsion engines:-

2 engine aircraft: 45% of total in fleet

3 engine aircraft: 40% of total in fleet

4 engine aircraft: 37.5% of total in fleet

Lift engines:-

6 engine aircraft: 16% of total in fleet

12 engine aircraft: 16% of total in fleet

Gearbox and rotor spares investment were taken as equivalent to 40% of the total in the fleet. This item would probably be held as components rather than complete units.

4. AMORTIZATION, INTEREST AND INSURANCE

The total investment per aircraft was assumed to depreciate to zero over 14 years, that is over 26400 flight hours. The investment was taken as the prime cost plus the proportion of the spares holding allocated to each aircraft.

Interest was taken as 5% of the investment per annum, and insurance as 2% of the investment per annum.

5. ENGINEERING

For the case of a one hour block time the total engineering labour and material costs was taken as:-

Airframe:- $\pounds(10+0.0013W_a)$ per hour

where W_a is the difference of the empty weight equipped and installed powerplant weights, lb.

Propulsion engines:- $\pounds(12+0.00062T)$ per hour, per engine
where T is the static thrust of each engine

Lift engines:- $\pounds(0.0009T)$ per hour, per engine

9000 H.P. shaft engines:- $\pounds 18$ per hour, per engine

Gearboxes and transmission:- $\pounds 24$ per hour for each gearbox unit.

Rotor system:- $\pounds 32$ per hour for each rotor unit.

Auto controls and APU:- $\pounds 14$ per hour, total.

In estimating the airframe engineering costs an attempt was made to make allowance for the more complex flap and flying control systems used in some of the designs. This was based on the work of Coughlin Ref.16. Consideration was also given to reduced undercarriage engineering costs with reduction of approach speed. It was found that the various effects tended to cancel apart from the case of the VTOL designs where a slight relative reduction could reasonably be anticipated. This amounted to less than 1% of the total direct operating costs and for

simplicity was neglected.

6. FUEL

Calculations were based on two fuel costs which are anticipated to cover the range likely to be experienced in the foreseeable future.

Fuel costs A:- 12p/gallon (1.5 p/lb)
Fuel costs B:- 36p/gallon (4.5 p/lb)

7. CREW

In each case allowance was made for two aircrew at £55/hour and four cabin staff at £30/hour, total.

8. LANDING AND NAVIGATION FEES

Landing fees were calculated as $£0.6 \times 10^{-3}W$

En route navigation fees were assumed to be
 $£1.5(W \times 10^{-3})^{0.5}$

APPENDIX C

NOISE

A simplified approach has been made to the problem of estimating the noise characteristics of the various designs. This was felt to be justified in view of the paucity of information in some cases and the prime requirement to establish relative rather than absolute values.

The method used was to estimate the sideline noise levels at 500 ft distance for both take off and landing conditions and use these to determine the area of the 80 PNdB ground footprint. This was done by assuming cylindrical noise fields defined relative to the ground plane by the climb out and approach angles. Where a significantly different noise level below the flight path was anticipated a correction was applied, and directivity was allowed for in the case of vertical takeoffs. In all cases the noise was assumed to be attenuated at 8 dB for each doubling of the distance from the 500 ft datum.

1. PROPULSION ENGINES

The take off noise level at 500 ft was estimated from the following formula, partly derived from published Rolls Royce data and quoted in Ref.17:-

$$\text{PNdB} = 10 \log \left[\text{Antilog}\{8.8 + \log R\} + \text{Antilog}\{11.5 - 1.7 \log R\} + \text{Antilog}\{12.8 - 3 \log R\} \right] + 10 \log \left(\frac{T}{80000} \right) - \Delta$$

The three terms in the square brackets represent, respectively, the compressor, turbine and exhaust noise.

R is the bypass ratio and T the static thrust in lbs.

Δ is a correction to allow for development and layout of the airframe/engine combination

Δ was assumed to be 1.5 dB for the basic versions of the powerplants, 5 dB when fully developed from a noise aspect and 7.5 dB when intake noise shielding was present. In all the cases where propulsion engine noise is important the bypass ratio assumed was 10 so that the noise equation reduces to:-

$$\text{PNdB} = 102.8 + 10 \log(T/80000) - \Delta$$

For the landing and other reduced thrust cases the take off noise level was reduced by :-

$$15 \left(\frac{T_R - T}{T} \right) \text{ dB}$$

where T_R is the reduced thrust.

2. FAN LIFT ENGINES

The same equation was used for the propulsion engines except that a bypass ratio of 12 was assumed and Δ was taken as 1.5 dB initially and 7.5 dB for the fully noise developed engines:-

$$\text{PNdB} = 101.9 + 10\log(T/80000) - \Delta$$

The directivity effect during vertical flight was allowed as in paragraph 6 for the tilt wing concept.

3. AIRFRAME

Using the little evidence available and comparing it with rotor broad band noise Ref.17 suggests that at 500 ft distance

$$\text{PNdB} = 10\log\left(\frac{W}{S}\right) + 10\log W + 20\log V - 31$$

where W is the weight, lbs, S the wing area in sq ft and V the velocity in ft/sec.

4. EXTERNALLY BLOWN FLAPS

The data used for estimating the noise of the externally blown flap aircraft was derived from Refs. 18, 19 and 20. Some difficulty was encountered in reconciling the various sets of information and because of the severity of the noise problem in this case the most optimistic assumptions were made. The exhaust velocity of the bypass ratio 10 engines was assumed to give an equivalent pressure ratio of between 1.3 and 1.5 over the flaps as a whole.

In the take off case, with the flaps set at 10° - 20° it would appear that the noise level at 500 ft for 80,000 lb total thrust will be of the order of 113 to 118 PNdB. This range covers the variation of pressure ratio and about 2 dB difference in scaling from the various references. It includes a subjectivity allowance of 8-9 dB, deduced from Ref.19 and applies immediately below the flight path. A noise level of 114 PNdB was therefore taken for this case with the sideline level reduced to 109 PNdB as suggested by Ref.18.

In the landing condition the flap setting is 20° - 40° and this by itself causes about 2-3 dB increase in noise level. However this is associated with a reduced thrust and the net result is that the noise level is approximately the same as during take off.

Ref. 19 suggests that use of a mixer nozzle should enable overall noise reduction of 5 dB to be achieved, and this has been assumed for the quiet externally blown flap design.

5. AUGMENTOR WING

The basic augmentor wing noise was calculated using Ref.18. The nozzle pressure ratio as designed is 1.9, and the take off flap angle 20° . Making an allowance of 9 dB for subjectivity the noise level at 500 ft below the flight path for 80,000 lb thrust is found to be 109 dB. The sideline noise is quoted as 5 dB less than this. As with the externally blown flap the extra noise of 3 to 4 dB due to deflecting the flap to the landing position of 50° is partly offset by the reduction of thrust. It is likely that the landing noise level will be up to about 1 dB higher than the take off value, but as this is considered to be less than the accuracy of the prediction, identical values have been assumed.

Ref.12 considers ways of improving the noise level of augmentor wings. It suggests that the use of acoustic lining on the internal flap surfaces together with a multielement nozzle and screech screens should enable a reduction of at least 14 dB to be obtained. Ref.18 quotes a reduction of 8 dB maximum for a particular duct lining alone. The quiet augmentor wing design was therefore based on a reduction of up to 14 dB relative to the basic values quoted above.

6. TILT WING AIRCRAFT

The noise of the tilt wing aircraft is assumed to be due to the shaft turbine engines and the rotors. It is possible to regard the engine/rotor system as a fan engine of large bypass ratio. Thus the basic shaft turbine noise can be evaluated from the formula quoted for propulsive engines by using only the first two terms in the square brackets and relating the power developed to equivalent thrust. If it is assumed that the 9000 HP shaft engine is equivalent to a bypass ratio 10 engine of 15000 lb thrust then the noise level at 500 ft is:-

$$PN_{dB} = 101 + 10 \log \left(\frac{SHP}{48000} \right) - \Delta$$

where SHP is the shaft horse power.

As far as the rotor is concerned it is assumed that the design is such that broadband noise is dominant. This is associated with a 7 blade rotor and tip speed of 750 ft/sec.

The noise evaluation was based on information contained in Ref.21, but modified in accordance with the suggestions of Ref.22. For vertical flight in zero wind conditions the noise 500 ft from the source in this case is about:-

$$\text{PNdB} = 60\log V_t + 20\log C_{L_t} + 10\log S_b + f(\theta) - 76$$

where V_t is the blade tip velocity, ft/sec, C_{L_t} is the tip lift coefficient, S_b is the total blade plan area, sq ft and $f(\theta)$ is the directivity factor. $f(\theta)$ is zero when the aircraft is vertically above the observer and -15 dB when it is alongside. For the blade characteristics of the design aircraft the noise level at 500 ft on the ground was estimated at 95 dB and 110 dB when 500 ft above the observer. The peak climb out level at the ground 500 ft from the take off point is 101 dB.

KEY TO TABLES 1 AND 2

- CTOL - Conventional take off and landing - above 4000 ft runway.
 - RTOL - Reduced take off and landing - no lift augmentation.
 - STOL - Short take off and landing - lift augmentation.
 - VTOL - Vertical take off and landing
 - RE - Rear fuselage mounted powerplants
(Powerplant noise shielding)
 - UW - Underwing powerplants
 - BF - Externally blown flap lift augmentation
 - FL - Fan lift engines for vertical thrust
 - AW - Augmentor wing system
 - QAW - Quiet augmentor wing
 - TW - Tilt wing concept (Twin rotors)
 - A - Costs with fuel at 1.5p/lb
 - B - Costs with fuel at 4.5p/lb.
 - Q - Quietened powerplants and lift augmentation
 - PNdB } Flyover noise level at 500 ft
 - PNdB } Sideline noise level at 500 ft
- (Landing and take off values similar)

TABLE 1

Summary of Aircraft Characteristics.

Class of aircraft	CTOL				RTOL		STOL						VTOL	
	Runway length - ft	7000 UW	5000 UW	5000 RE	4000 UW	4000 RE	2600 BF	2000 BF	2000 FL	1600AW	1600 QAW	0 FL	0 TW	
Gross weight lb		82000	87300	89800	97500	101500	111200	118500	113500	121400	126800	125300	120000	
Wing area sq ft		680	970	1000	1160	1200	1510	1600	1345	1640	1710	1000	1330	
Wing loading, take off lb/sq ft		120	90	90	84	84	74	74	84	84	84	125	90	
Wing loading, landing lb/sq ft		110	83	83	77	77	64	64	77	77	77	120	88	
Number of propulsion engines		2	2	2	2	3	4	4	3	4	4	2	4	
Static thrust of each prop. engine lb		11500	12200	12600	22600	15600	14000	20400	15600	20600	21500	21500	9000 HP	
Number of lift engines									6			12		
Thrust of each lift engine lb									8300			14500		
Thrust/weight ratio, static		0.28	0.28	0.28	0.46	0.46	0.50	0.68	0.41	0.68	0.68	0.34	0	
Take off safety speed, knots		145	130	130	122	122	108	100	86	93	93	0	0	
Lift off lift coefficient		1.5	1.5	1.5	1.62	1.62	3.0	3.0	1.6	2.5	2.5	0	0	
Climbout angle, degrees		6.5	6.5	6.5	10	10	6	14.5	14	14.5	14.5	90	90	
Approach speed, knots		135	114	114	104	104	79	79	79	72	72	0	0	
Approach lift coefficient		1.7	1.7	1.7	2.1	2.1	3.4	3.4	2.1	3.6	3.6	90	90	
Approach angle, degrees		3	3	3	5.5	5.5	7.5	7.5	7.5	7.5	7.5			
First cost fm		2.41	2.64	2.74	3.08	3.24		3.89	3.76	4.01		4.5	4.3	
Investment fm		2.85	3.12	3.24	3.62	3.81		4.82	4.55	4.95		5.44	5.8	
Basic cost f/hour		194	213	222	247	250		327	311	336		378	404	
Engineering cost f/hour		116	122	126	151	156		194	206	197		290	273	
Fuel cost f/hour - A; 1.5p/lb		67	71	76	79	84		106	114	108		174	76	
B; 4.5p/lb		201	213	228	237	252		318	342	324		522	228	
Total cost f/hour - A		525	557	577	636	660		799	802	816		1019	926	
- B		659	699	729	794	828		1011	1030	1032		1367	1078	
Cost p/pas.mile - A		1.62	1.72	1.78	1.96	2.04		2.46	2.48	2.5		2.6	2.96	
- B		2.03	2.16	2.25	2.45	2.56		3.12	3.18	3.19		3.49	3.33	
Noise level, take off PNDdB		96	96		99			114	101.5	109		111.5		
landing PNDdB		87	87		88			109	98.5	104		109.5		
80 PNDdB footprint, sq miles		3.6	3.4		3.1			29.5	5.4	7.9		9.4		
Noise level, take off PNDdB-Q		92.5	92.5		95			109	97			106	102	
landing PNDdB-Q		85	85		87			104	93.5			104	101	
80 PNDdB footprint, sq.miles-Q		2.3	2.1		2.0			10.7	2.33			2.5	1.7	

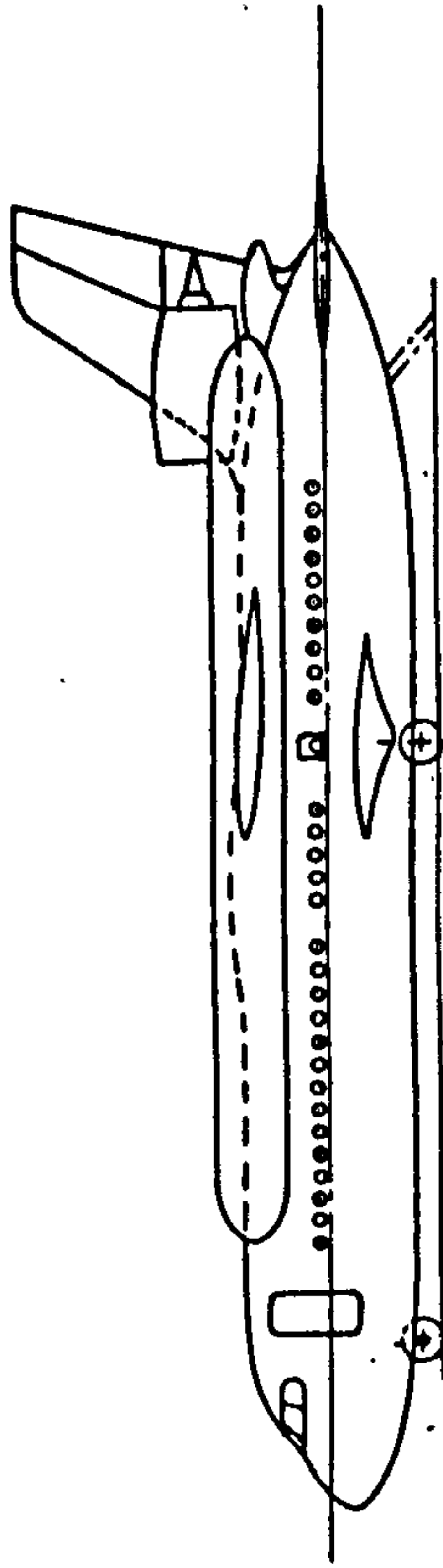
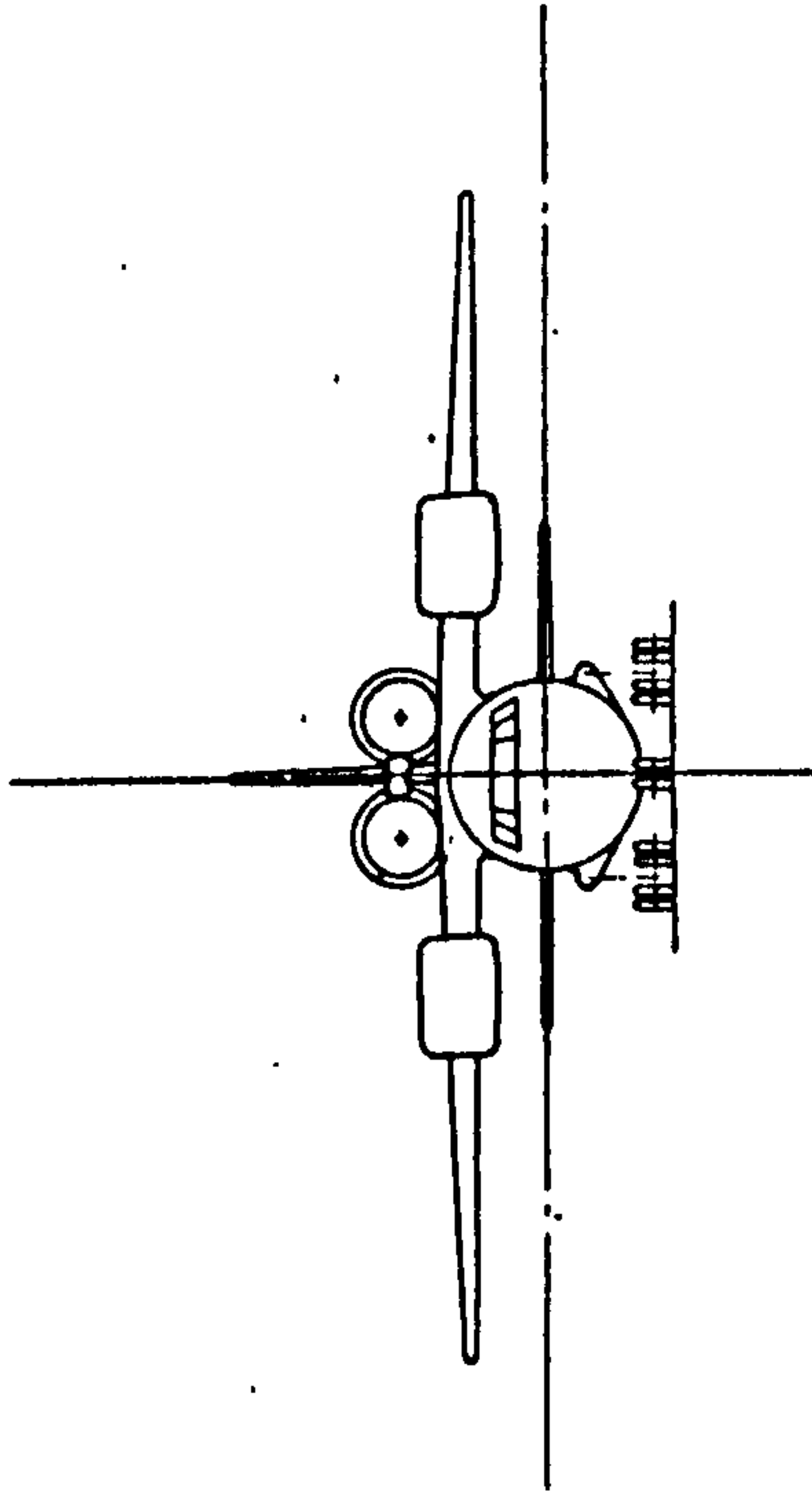
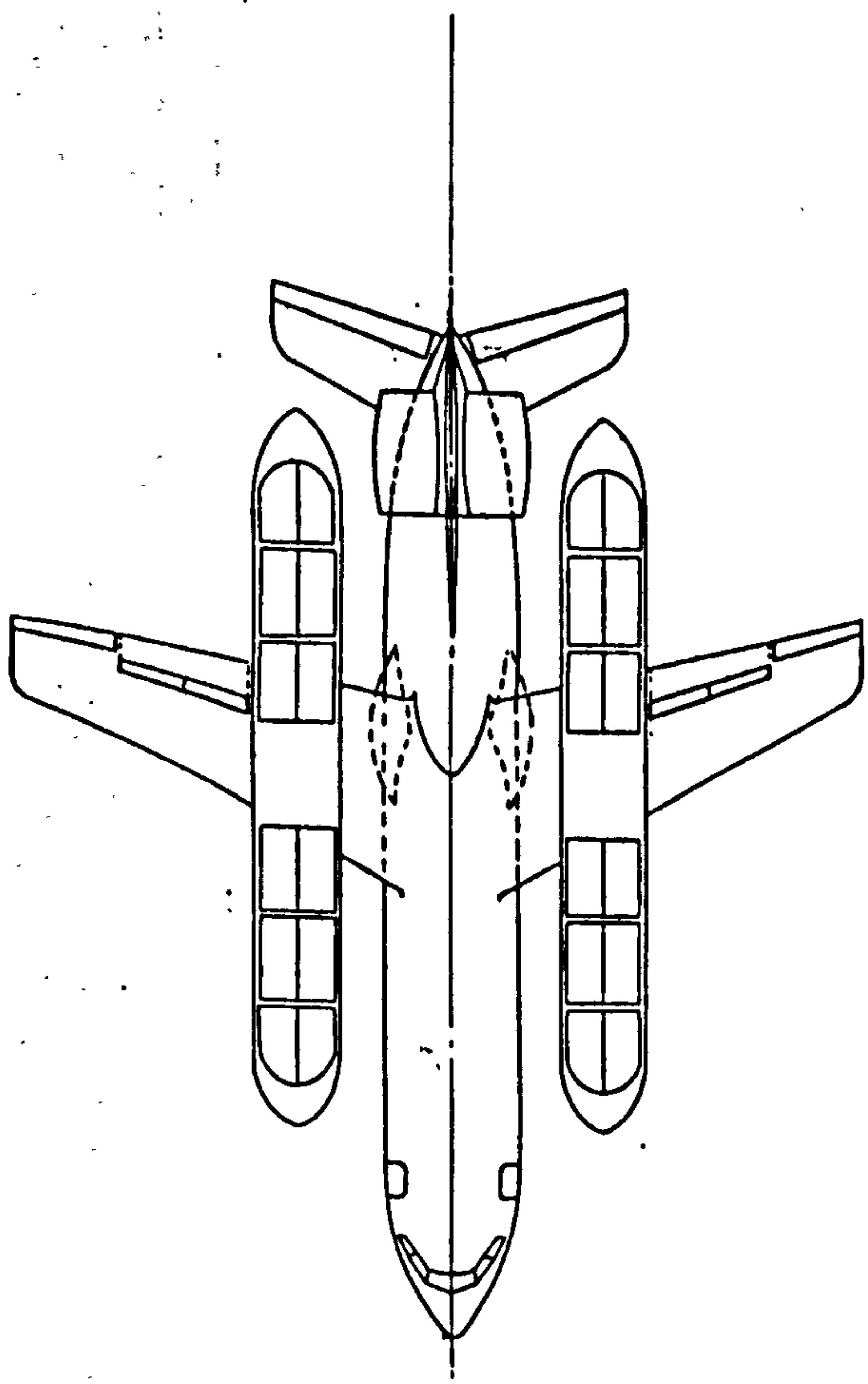
See Key for explanatory note

TABLE 2

Summary of Aircraft Weight Breakdowns

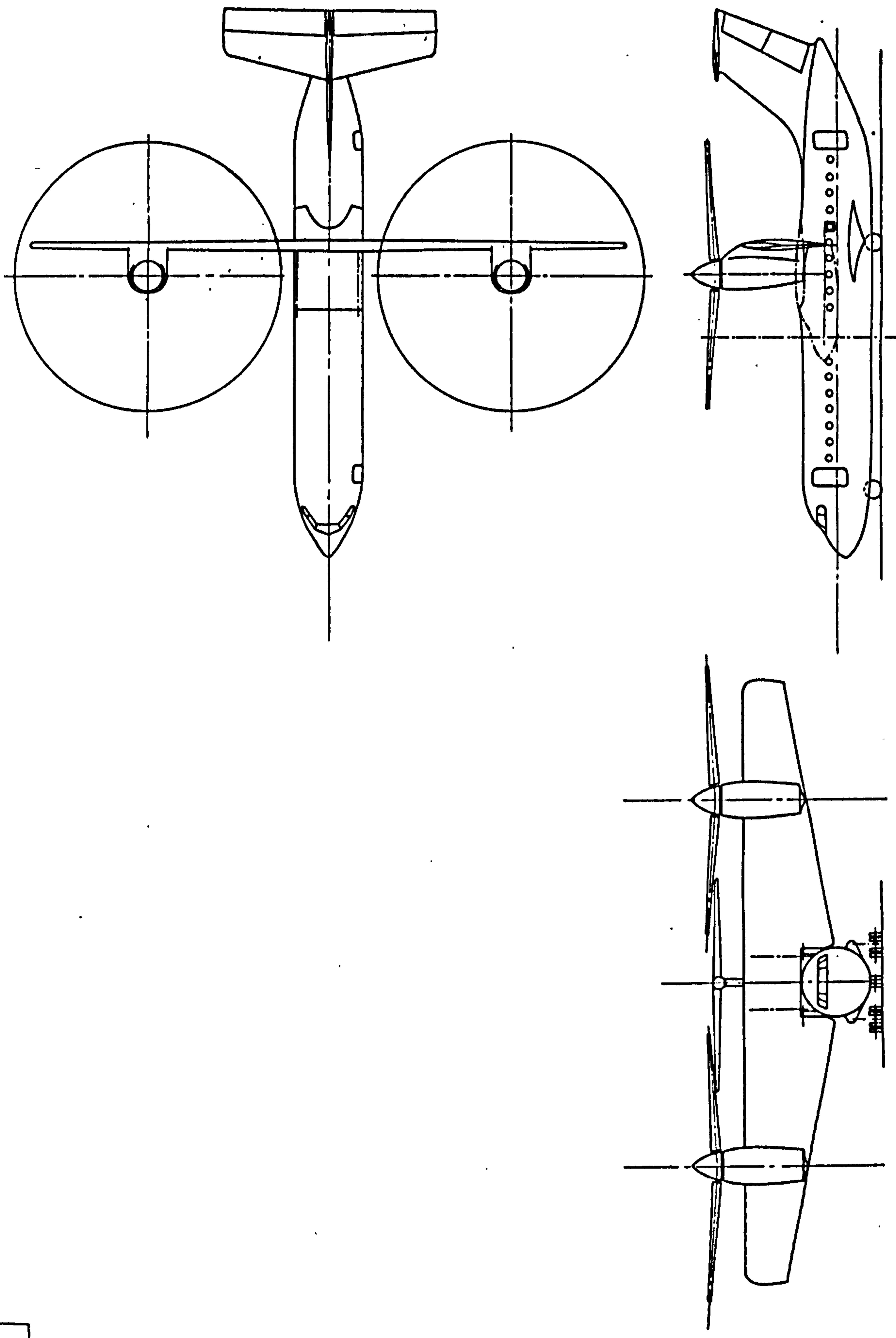
Class of aircraft	CTOL			RTOL		STOL							VTOL	
	Runway length - ft	7000 TW	5000 TW	5000 RE	4000 TW	4000 RE	2600 BF	2000 BF	2000 FL	1600 AW	1600 QAW	0 FL	0 FL	0 TW
Fuselage		10400	10400	11500	10400	11500	10400	10400	11500	10400	10400	11500	11500	11250
Wing (less flaps)		4200	6900	7800	7200	8200	9600	9900	9300	10400	10870	9650	9650	10500
Flap system		1450	1900	1900	2500	2700	2400	2600	2700	4200	4390	1500	1500	2300
Tail unit		1830	2420	2480	3000	3130	3940	4060	3140	4200	4350	3500	3500	3600
Undercarriage		2970	3170	2970	4220	4470	5400	5590	4900	4670	4880	3390	3390	3400
Nacelle structure and pylons		200	200	200	300	400	400	400	2000	400	400	5960	5960	2300
Total Structure		21050	24990	26850	27620	30400	32140	32950	33540	34270	35290	35500	35500	33350
Propulsion engine installation		6300	6700	6900	12500	12900	16900	22500	12900	22700	23700	8400	13120	10900
Lift engine installation/rotors									3800					8700
Total Powerplant		6300	6700	6900	12500	12900	16900	22500	16700	22700	23700	21520	21520	19600
Fuel system		1000	1000	1000	1100	1100	1200	1240	1300	1240	1290	1300	1300	1250
Flying controls, auto controls		1140	1220	1260	1440	1520	1600	1670	1900	1690	1760	2000	2000	2950
Air ducting/transmission							400	400		1320	3520			9600
Power supplies		4000	4000	4000	4000	4000	4000	4000	4000	4000	4000	4000	4000	3600
Auxiliary power unit		530	530	530	530	530	530	530	530	530	530	530	530	
Radio and radar		1500	1500	1500	1500	1500	1500	1500	1500	1500	1500	1500	1500	1500
Instruments		600	600	600	600	600	600	600	600	600	600	600	600	600
Total Systems and Installation		8770	8850	8890	9170	9250	9830	9940	9830	10880	13200	9930	9930	19500
Air conditioning and de-icing		2330	2400	2400	2500	2500	2330	2600	2500	2600	2600	2330	2330	2300
Furnishings, cabin service		7450	7450	7450	7450	7450	7450	7450	7650	7450	7450	7650	7650	7600
Passenger service disposables		1250	1250	1250	1250	1250	1250	1250	1250	1250	1250	1250	1250	1200
Crew		1100	1100	1100	1100	1100	1100	1100	1100	1100	1100	1100	1100	1100
As Prepared for Service		48250	52740	54840	61590	64850	71000	77790	72570	80250	84590	79280	79280	84650
Passengers, 108		21600	21600	21600	21600	21600	21600	21600	21600	21600	21600	21600	21600	21600
Fuel, basic		12150	12960	13360	14310	15050	18600	19110	19330	19550	20610	24420	24420	13750
Take off Weight		82000	87300	89800	97500	101500	111200	118500	113500	121400	126800	125300	125300	120000

See Key for explanatory note



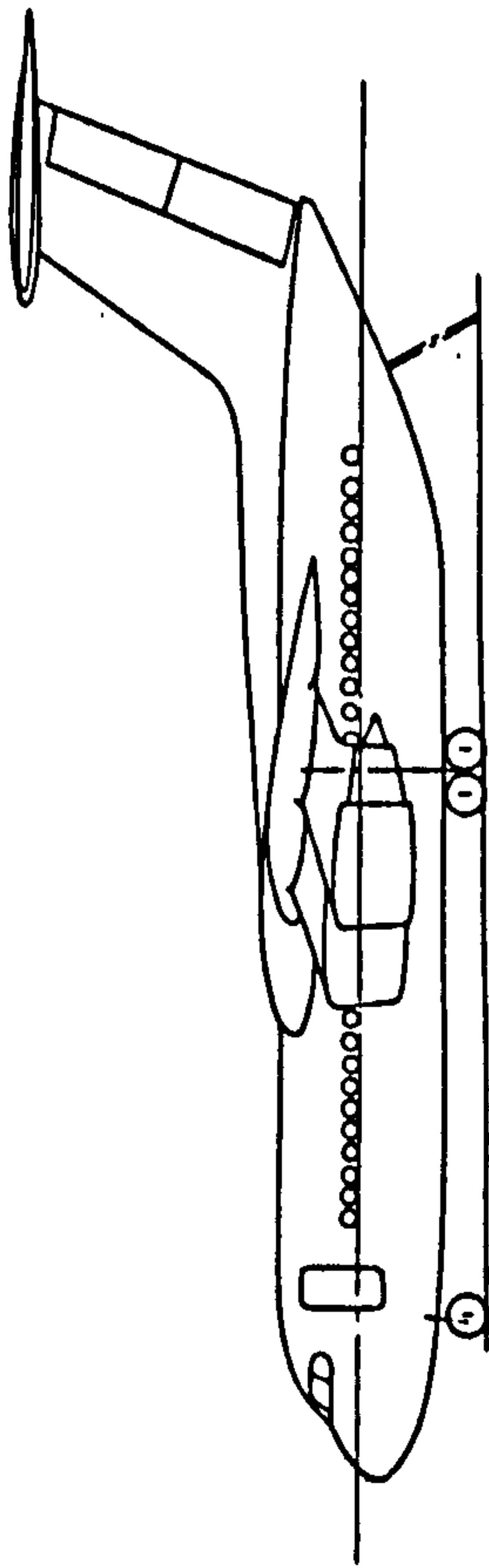
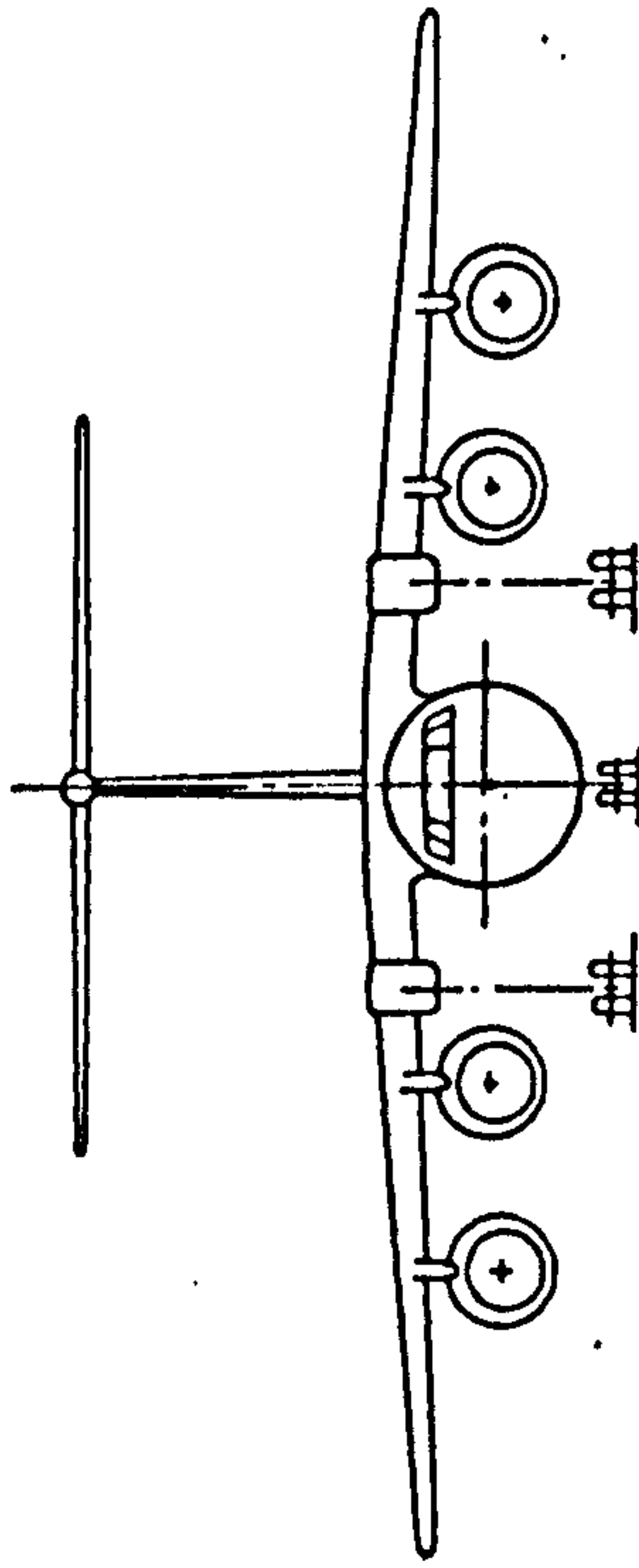
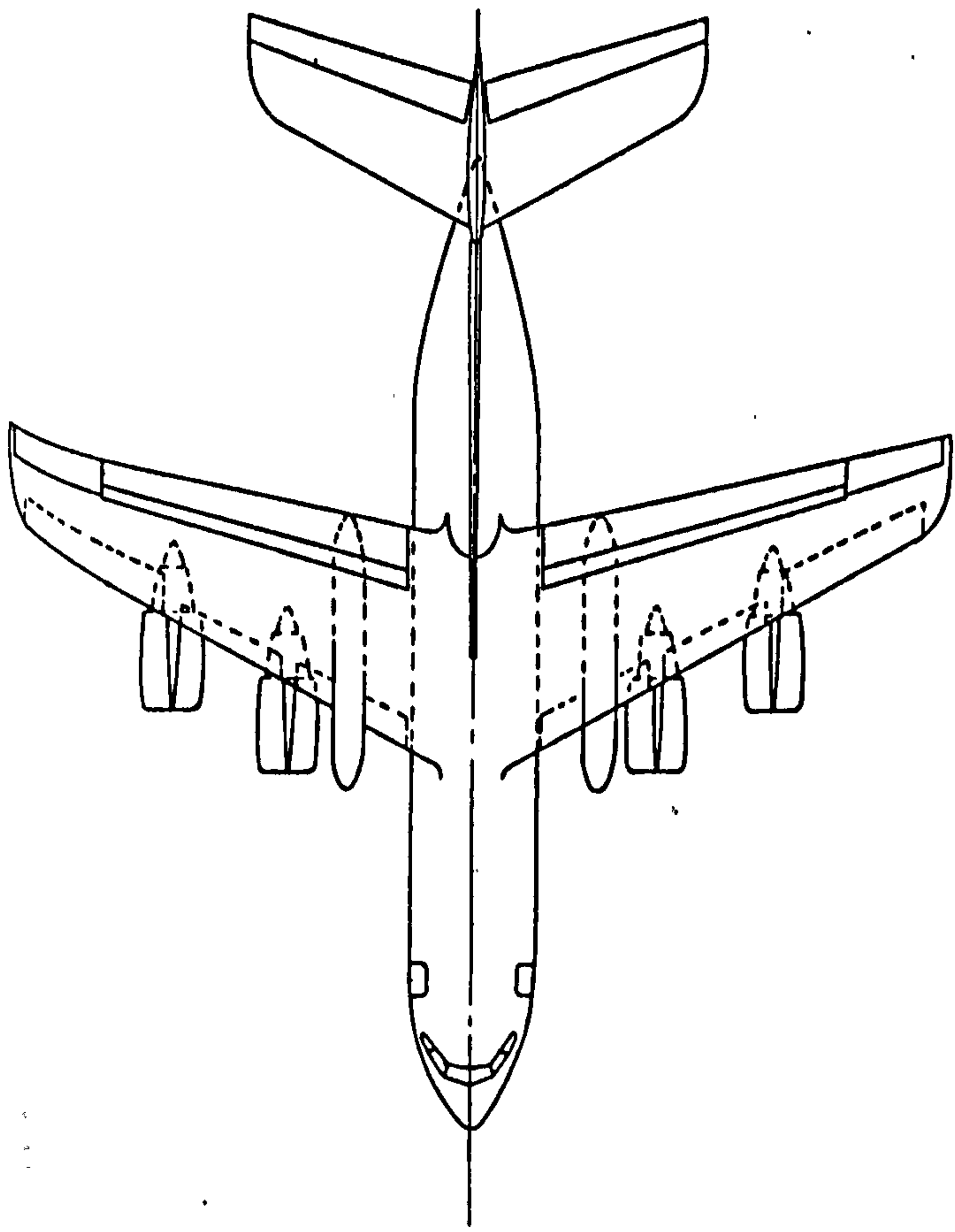
TITLE- A70 DESIGN PROJECT			
GENERAL ARRANGEMENT			
DRAWN	DESIGNED BY	A70-1	
J. H. H. H.			
2/14/70	DATE	OF	SHEETS

FIG. 1. GENERAL ARRANGEMENT OF A70 FAN-LIFT VTOL.



TITLE- 1973 PROJECT			
GENERAL ARRANGEMENT			
SCALE -	ISSUE -		R 73-1
DRAWN:	DATE - 16/7/73		

FIG. 2. GENERAL ARRANGEMENT OF R73 TILT WING VTOL



TITLE- 1971 DESIGN PROJECT			
GENERAL ARRANGEMENT			
SCALE- 1/100	ISSUE- 1		
DRAWN- S. H. H.	A 71 -1		
DATE- 28/7/71			

FIG. 3. GENERAL ARRANGEMENT OF A71 EXTERNALLY BLOWN FLAP STOL.

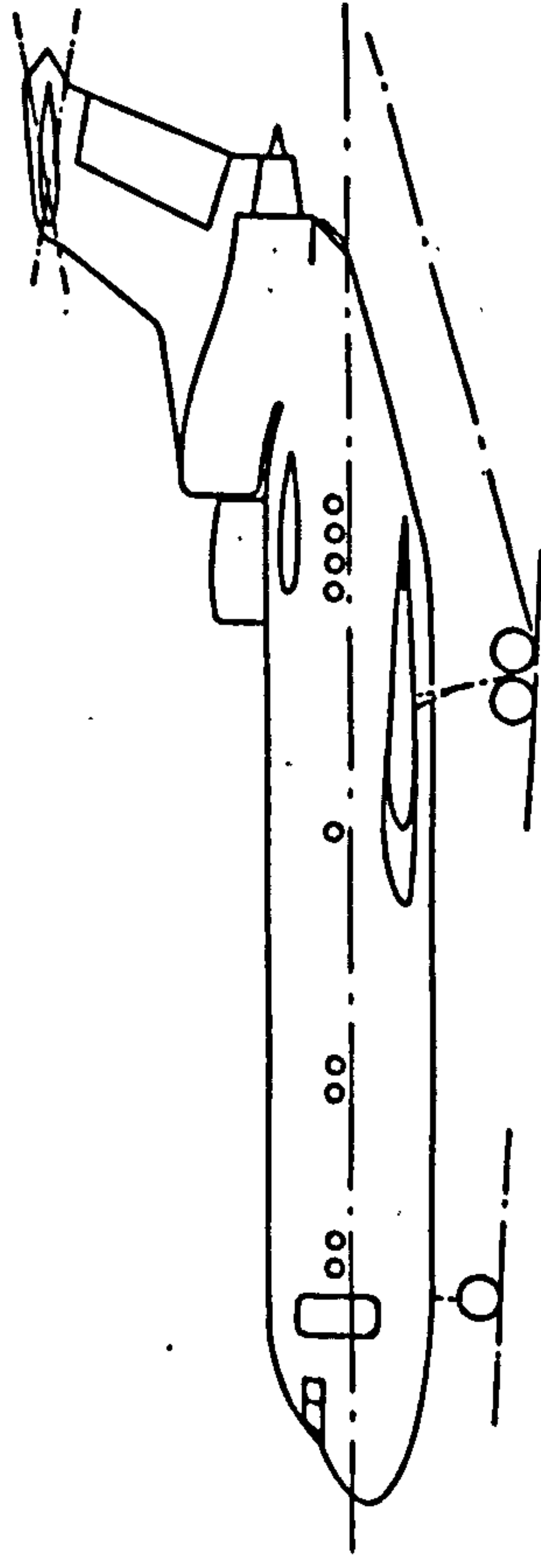
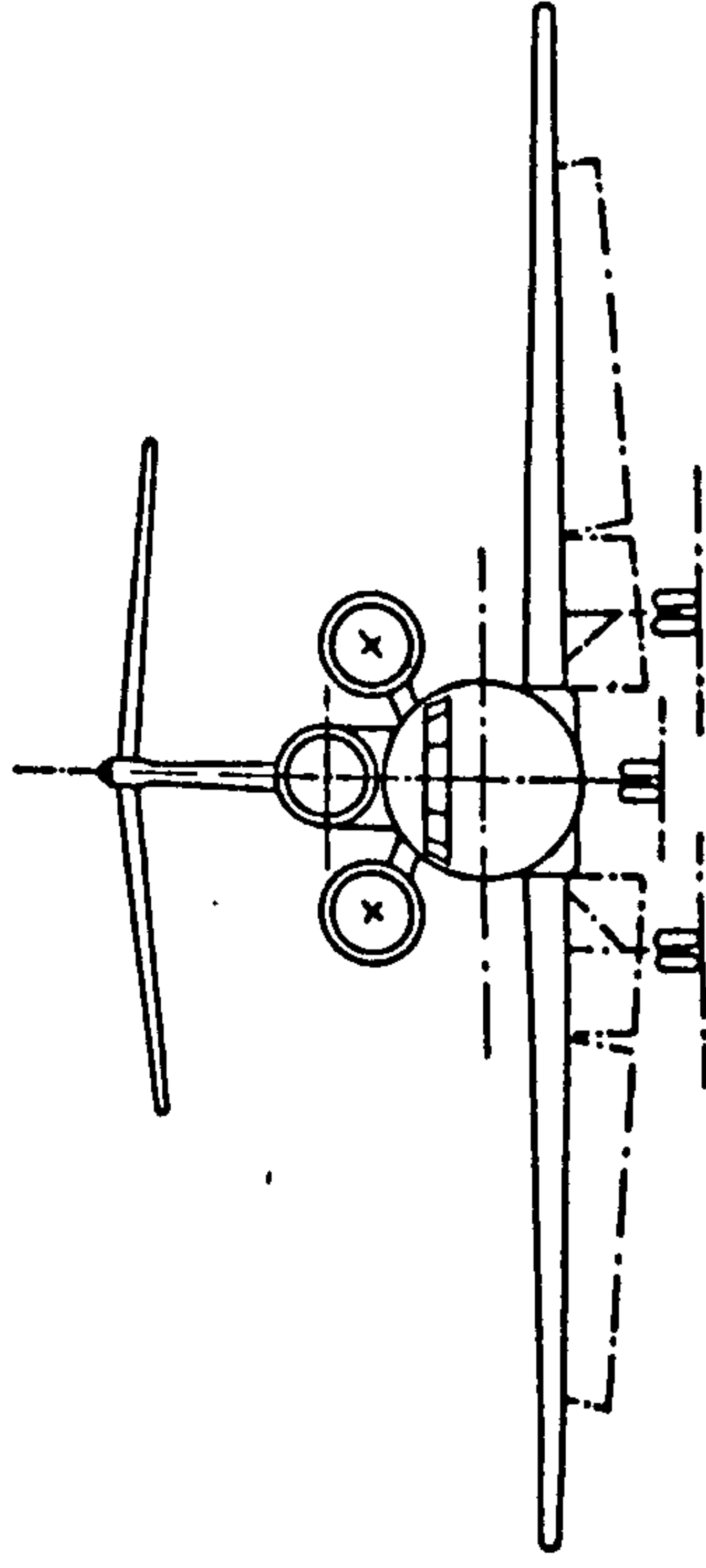
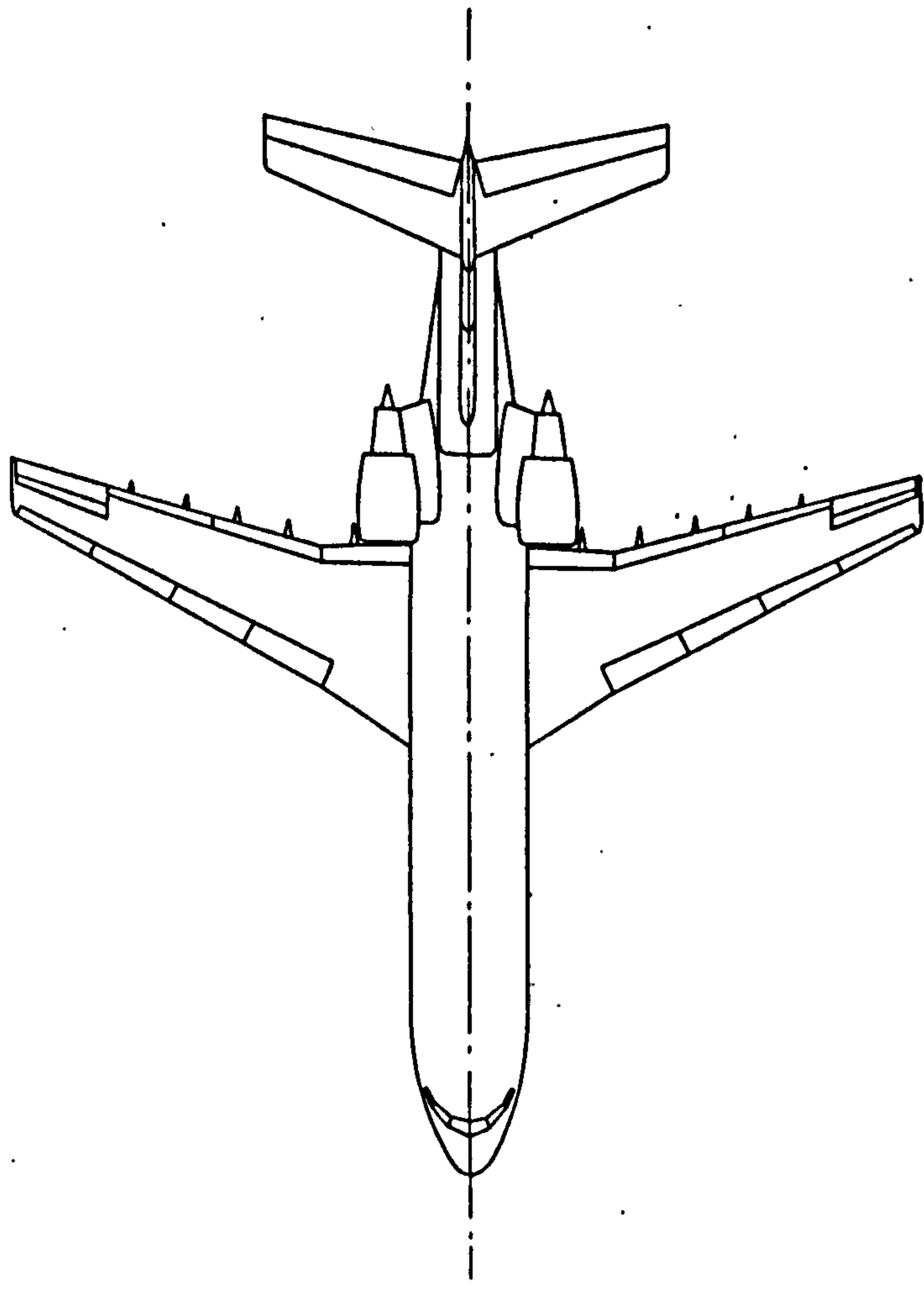


FIG 4. GENERAL ARRANGEMENT OF REAR ENGINE RTOL.

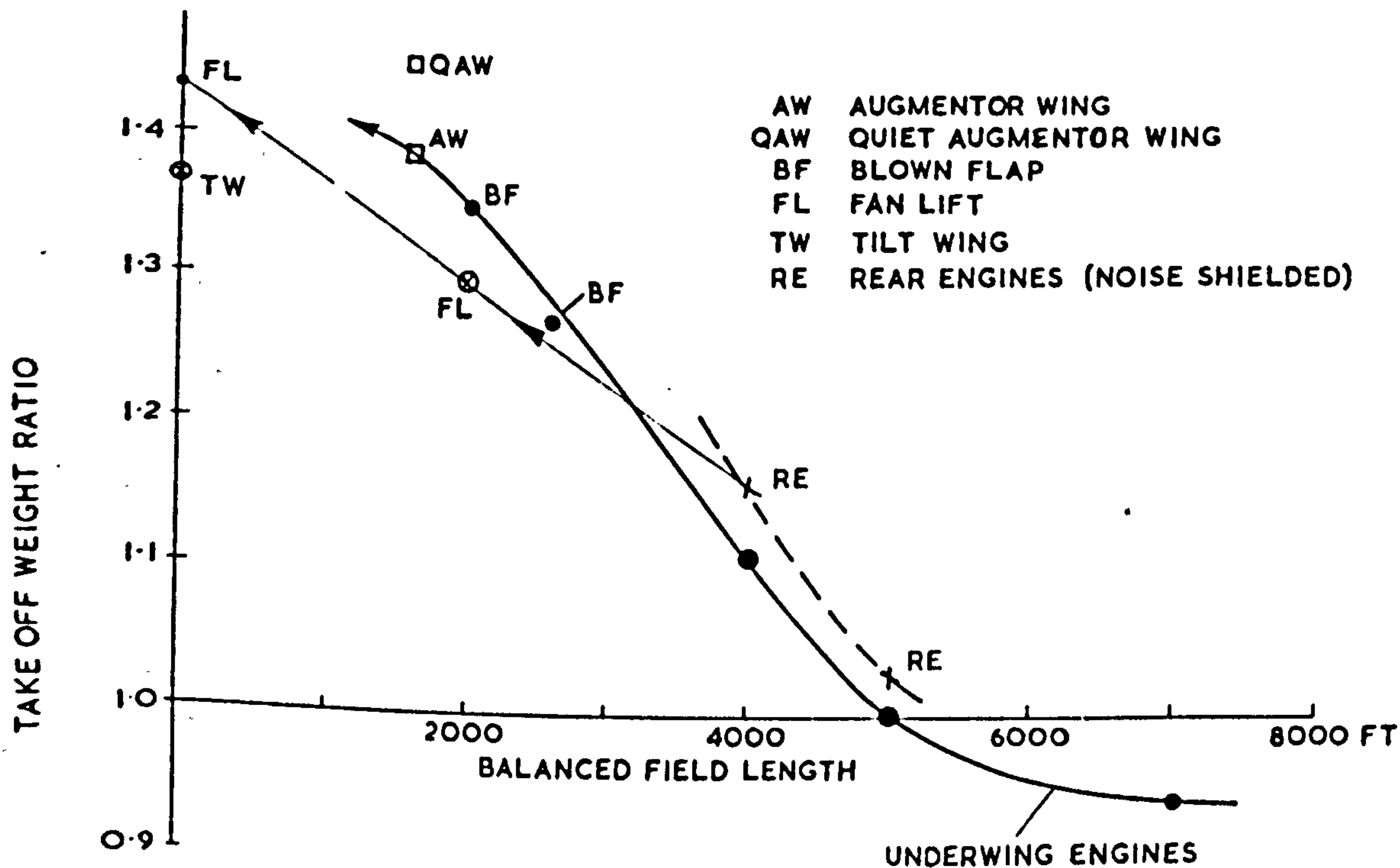


FIG. 5. RELATIVE GROSS WEIGHT.

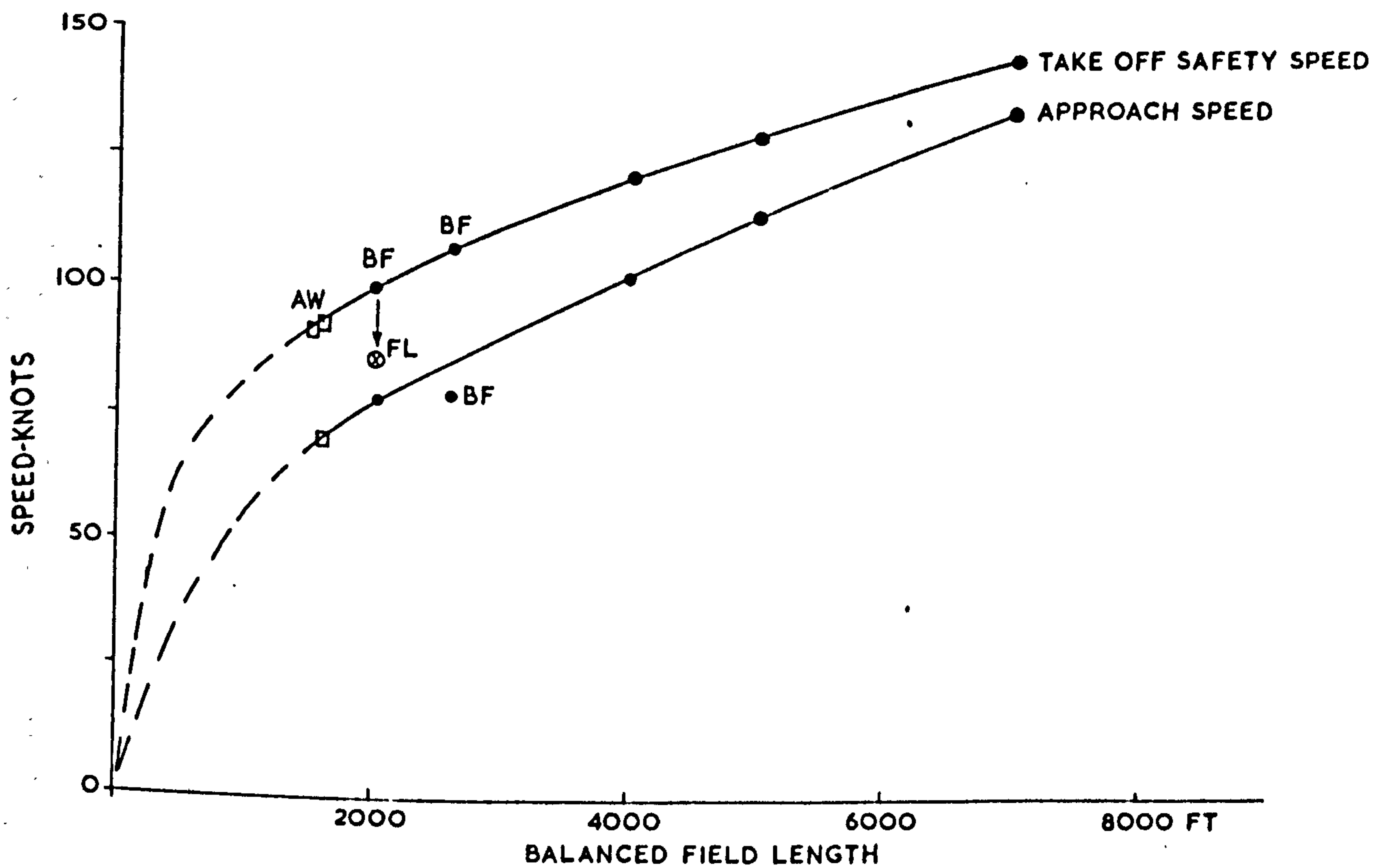


FIG. 6. TAKE OFF SAFETY AND APPROACH SPEEDS

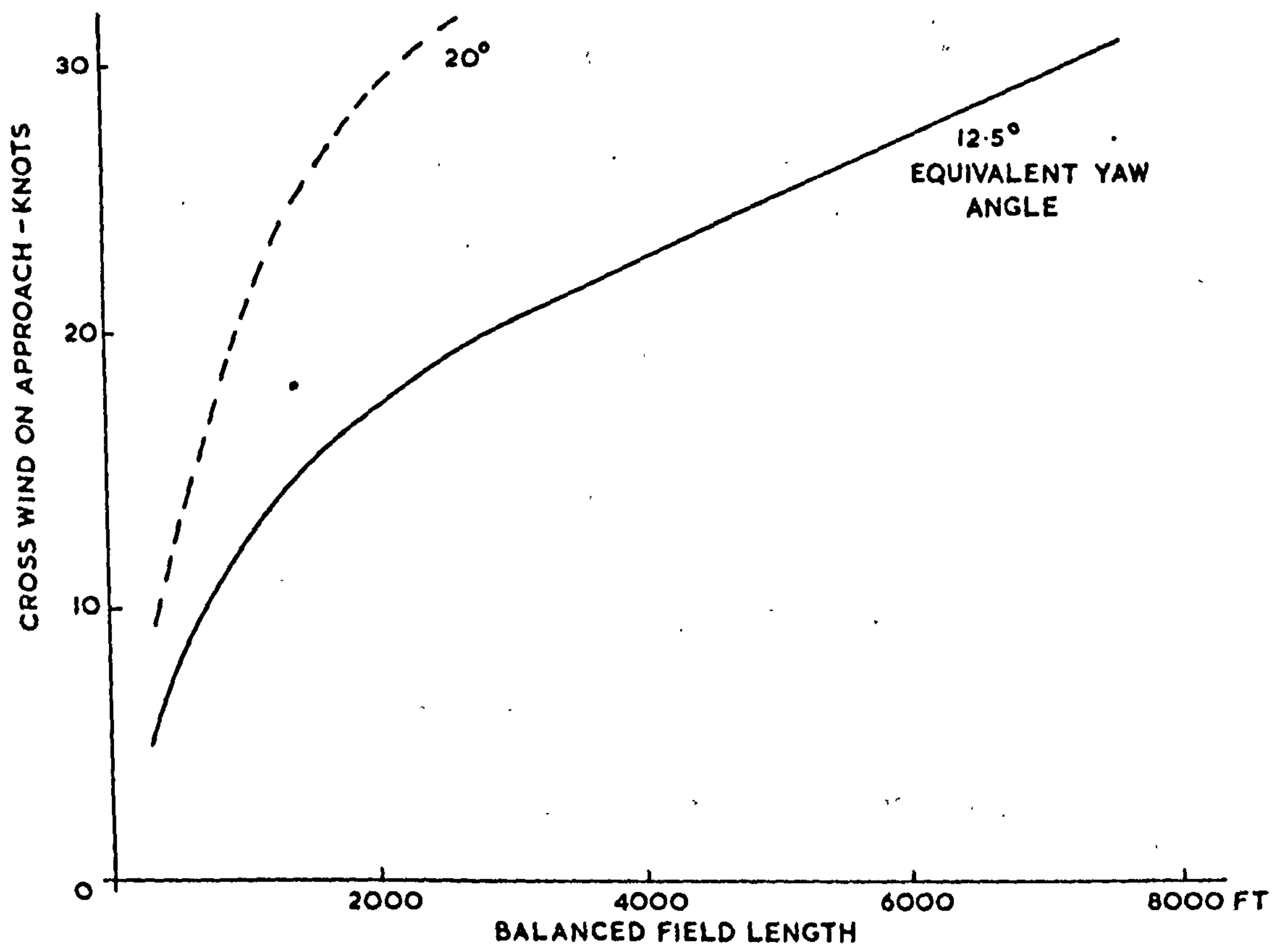


FIG. 7. CROSS WIND APPROACH CONDITIONS.

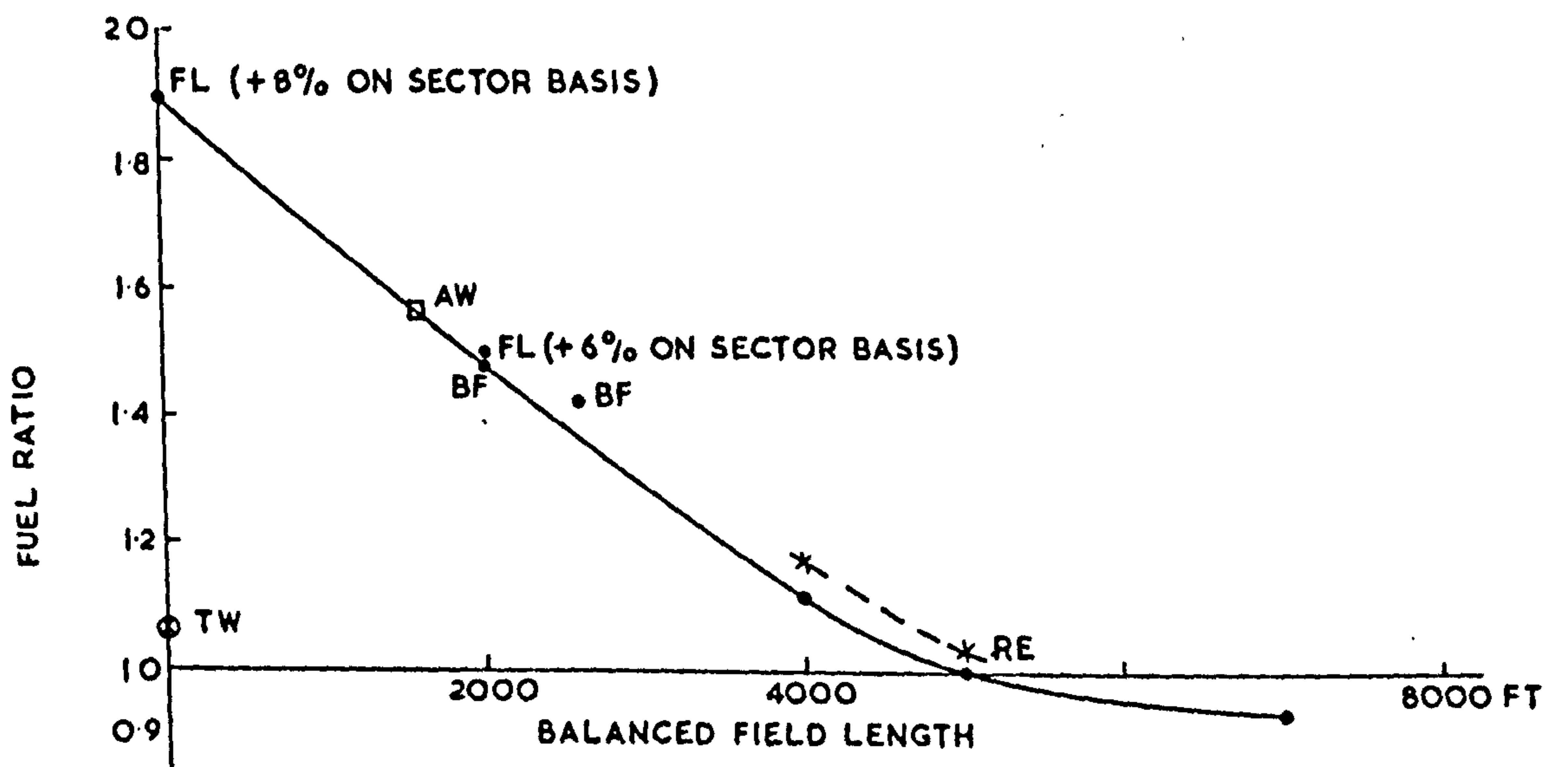


FIG. 8. RELATIVE FUEL LOADS CARRIED.

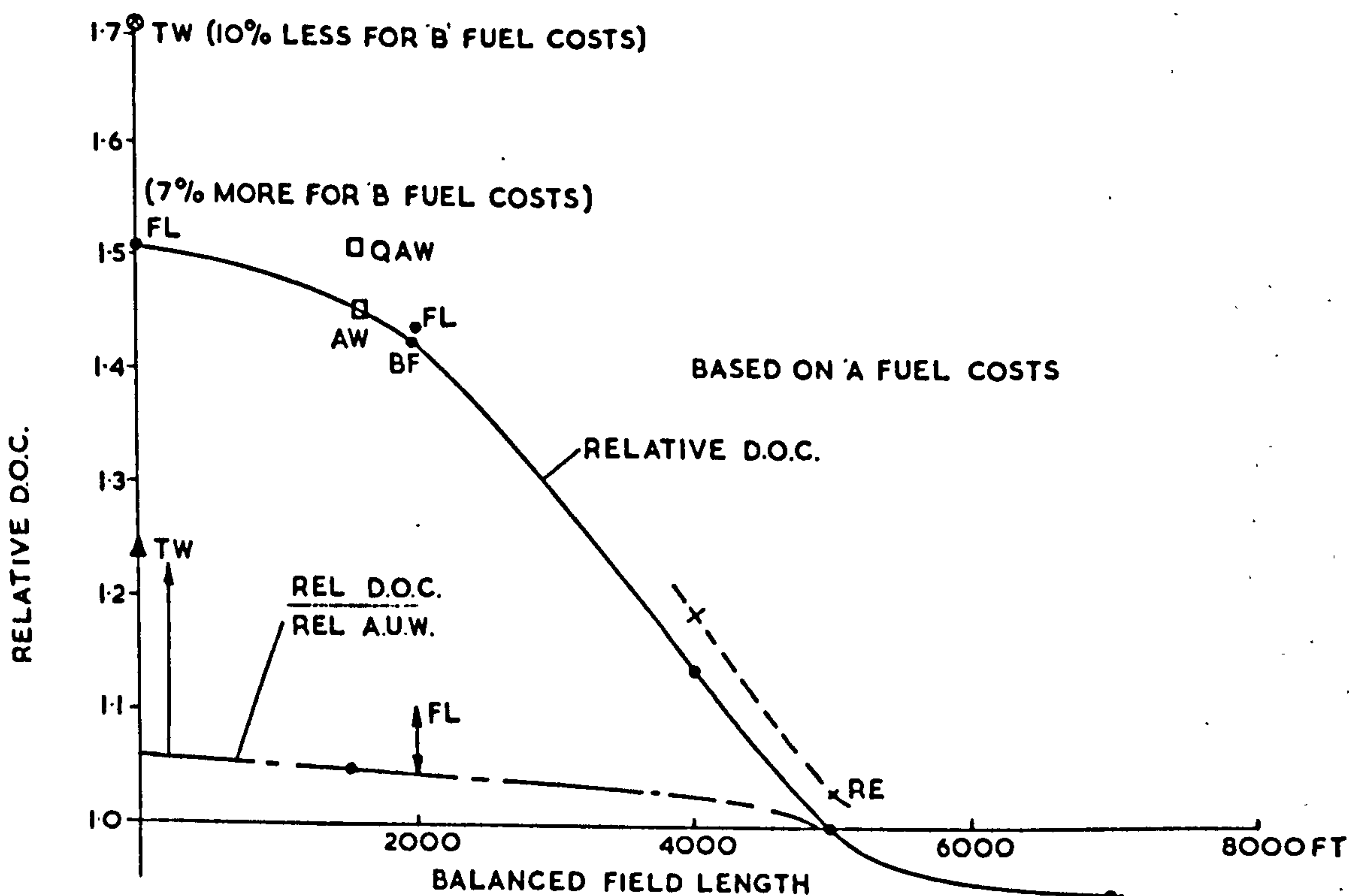


FIG. 9. DIRECT OPERATING COSTS COMPARISON

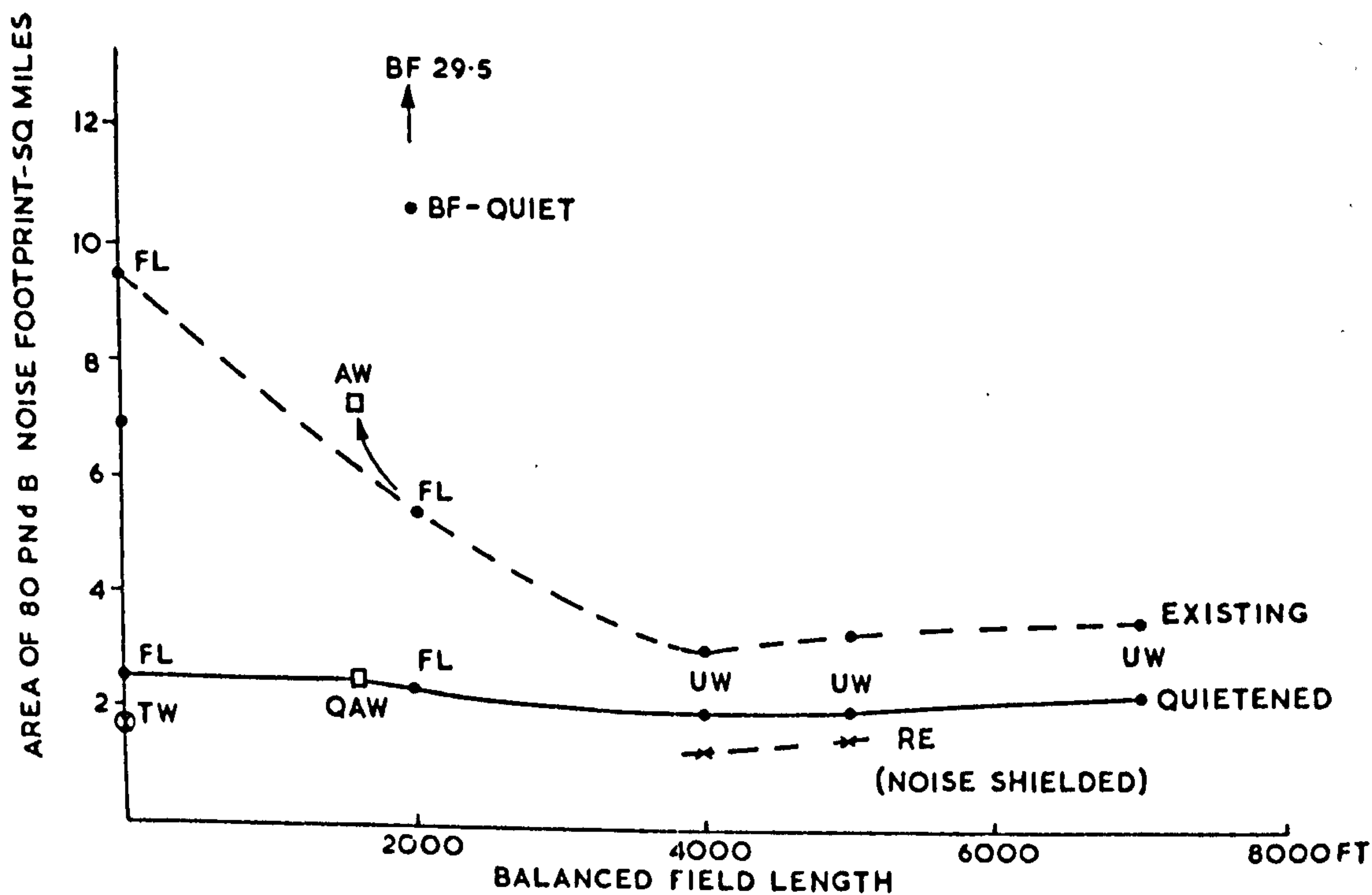


FIG. 10. NOISE FOOTPRINTS.

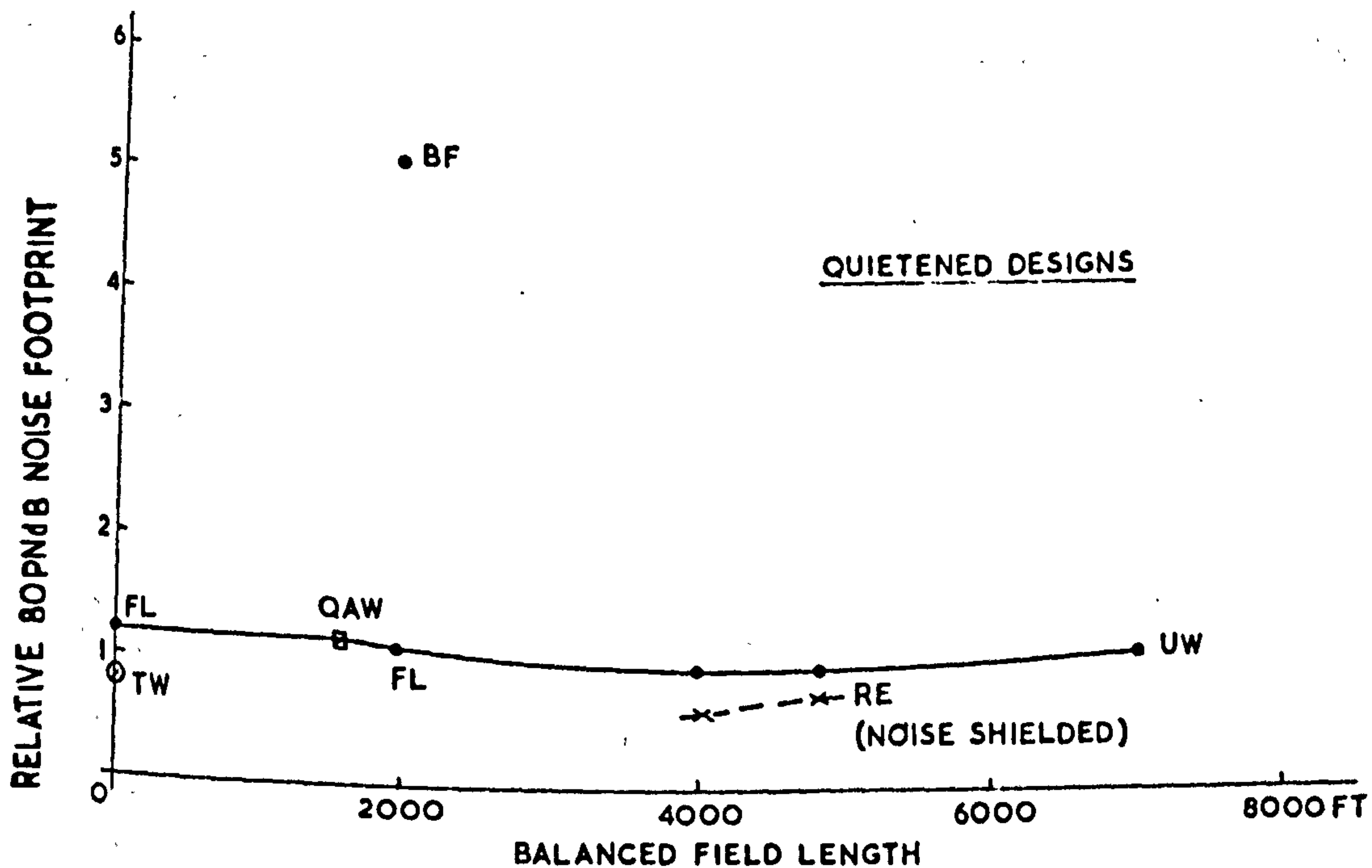


FIG. 11. RELATIVE NOISE FOOTPRINTS.

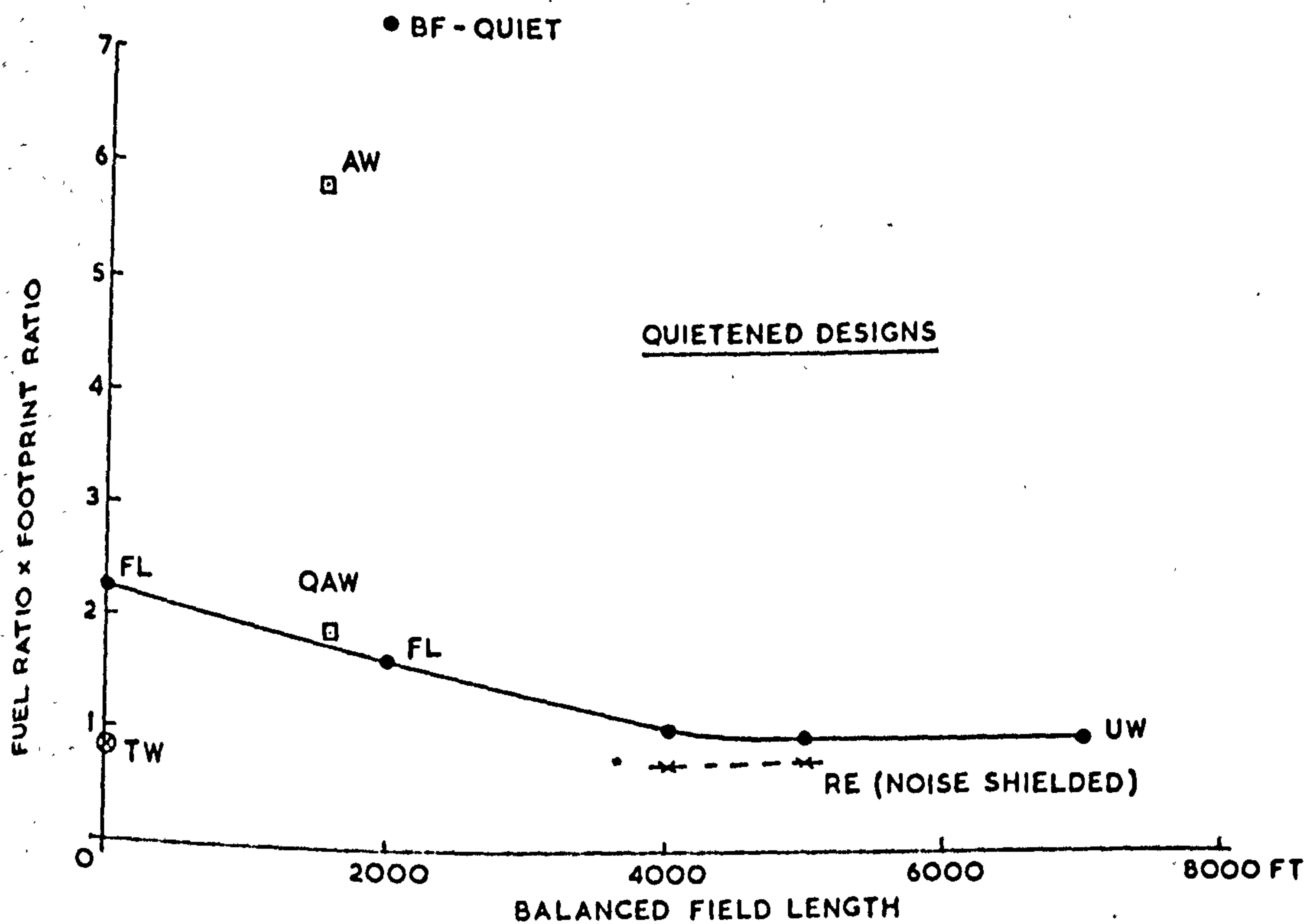


FIG. 12. ENVIRONMENTAL MERIT INDEX.

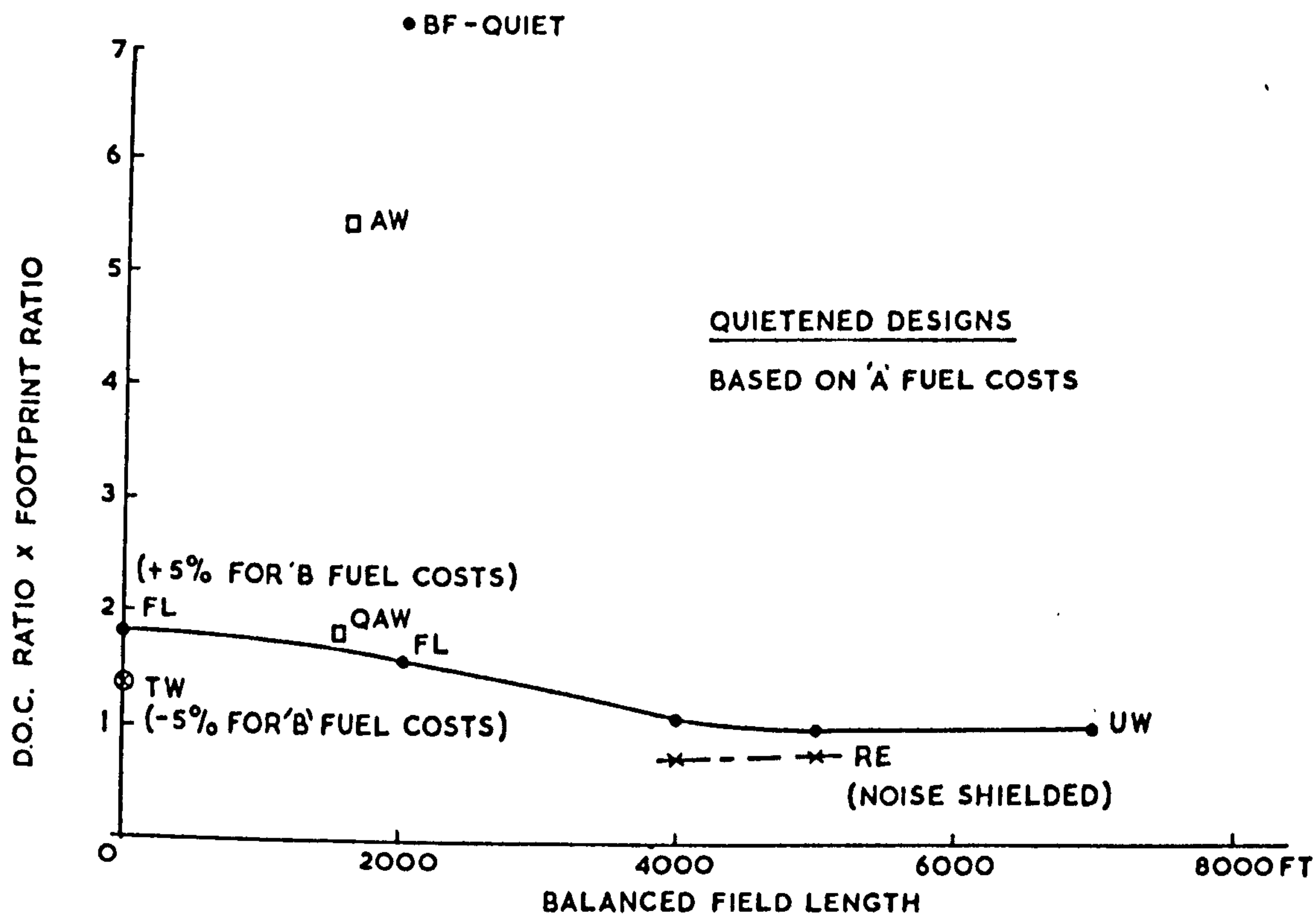


FIG. 13. ECONOMIC MERIT INDEX.

Cranfield Institute of Technology
College of Aeronautics

Aeroplane Design Study
STOL Airliner (A71)

Part 1 - Configuration description and data

by D. Howe and R.E. Ward

Summary

The interest in STOL airliners was reflected in the choice of a 100-118 passenger short range aircraft of this type as the 1971 design project. In addition to the use of the study for detailed investigation by the students of Aircraft Design it also served as the basis for an investigation of the low speed lift and control problems of STOL aircraft.

This report is concerned with a description of the configuration adopted and specification of geometric and aerodynamic data. As such it is the first part of the complete reporting of the investigation, subsequent parts being concerned with the more detailed work.

The aircraft was designed to operate from 2000 ft long single runways and have a cruising speed of up to $M = 0.83$ at 30,000 ft altitude. The estimated gross weight is 115,000 lb and when landing at 100,000 lb weight the approach speed is 79 knots. The high lift coefficients necessitated by this are obtained either by externally blown jet flaps or an augments wing arrangement.

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Notation

$a_1, a_{1T}, a_{1F}, a_{1BT}$	Lift curve slopes, per radian for wing, tailplane net fin and fin with body and tile effects respectively.
a_{2T}, a_{2F}	Lift curve slopes, per radian, due to elevator and rudder deflection, respectively.
b_1, b_{1T}, b_{1F}	Hinge moment coefficient slopes, per radian, due to wing, tail and fin incidence respectively.
b_2, b_{2T}, b_{2F}	Hinge moment coefficient slopes, per radian, due to aileron, elevator and rudder deflections, respectively.
\bar{c}	Mean wing chord (standard)
C_D	Drag coefficient
$(C_D)_{\mu=0}$	Low speed drag coefficient with $C_\mu = 0$
C_{FA}	low speed axial force coefficient
C_L	Lift coefficient
C_{M_0}	Pitching moment coefficient at zero lift
C_M	Increment to pitching moment coefficient due to lift at low speed with flaps deployed.
C_μ	Engine exhaust mass flow coefficient
M	Mach number
α	Fuselage datum angle of attack, degrees
	Non-dimensional stability and control derivatives:-
l_i, n_i, y_i	rolling moment, yawing moment and sideforce derivatives due to i given by:
	rolling p
	yawing r
	sideforce v
	rudder deflection ζ
	aileron deflection ξ

1. Introduction

The widespread interest in short take off and landing airliners is reflected in the choice of subject for the A71 design project. This study is concerned with an STOL short range jet airliner. For the purpose of the investigation STOL is defined as the ability to operate from single 2000 ft long runways. Whilst in some respects this choice of runway length is arbitrary it does coincide with the tentative requirements of certain operators. A greater runway length may be acceptable and could result in a more straightforward design but this is irrelevant in the present context as the aim of the study is to investigate the problems associated with a true STOL airliner.

There are two distinct aspects of the investigation. Firstly the A71 is the subject of the annual design exercise undertaken by the students of Aircraft Design and therefore the structural and mechanical features of the design are being examined in depth. Secondly it is a convenient vehicle on which to base a study of the low speed lift and control problems of STOL jet transports.

The payload-range and cruise speed performance have been chosen to be similar to that of the present generation of twin-jet airliners and also to that of the A70 lift fan VTOL airliner study.⁽¹⁾ This similarity of performance enables direct comparisons to be made between the various concepts.

For convenience the report of the investigation has been divided into separate parts. Part one is concerned with a description of the basic configuration and the overall data applicable to the aircraft. Subsequent parts will cover the detailed investigations.

2. High lift systems and powerplants

Two alternative means of developing the high lift coefficients required for low speed flight are being considered. Typically the approach lift coefficient must exceed 3 corresponding to a wing loading of approximately 70 lb/sq ft.

2.1 External flap blowing

The major study is based on the use of external flap blowing. The exhaust from four wing mounted Rolls Royce RB 410 fan engines is directed on to the lower surface of the double slotted trailing edge flaps. Each powerplant has a nominal static thrust rating of 14500 lb, and a bypass ratio of rather more than ten. The high bypass ratio has been chosen primarily to reduce the overall noise level, but the reduction of average efflux velocity and temperature also facilitates flap structural design. The downward turning of the exhaust by the trailing edge flaps is assisted by thrust deflectors which are located along the lower edges of the fan duct exits. These deflections enable the bypass flow to be directed upwards towards the knee of the flaps and this has the effect of increasing the angle through which the exhaust is turned. Full span leading edge flaps are used in conjunction with the deflectors and trailing edge devices.

The fans of the RB410 have variable pitch blades and are driven through gearboxes.

With this type of high lift system the failure of a powerplant has unusually serious consequences. Apart from the normal loss of thrust and the directional control problem there is also a significant loss of lift and an associated induced rolling moment. This introduces severe control problems which it is desirable to minimise. One possible way of doing this is to mechanically connect the adjacent fans on each side of the aircraft through the existing gearboxes. Providing a freewheel is incorporated in the drive the effect of a gas generator failure is considerably reduced. There is, of course, a substantial weight penalty and the effect of fan failure is not overcome. The possibility of fan failure due to foreign object ingestion or pitch control system faults is a matter of design requirement but the mechanical aspects of such an engine interconnection are considered to be worthy of investigation.

2.2 Augmenter wing

The alternative lift system is the use of an internally blown augmenter wing arrangement. In this case the powerplants are four Rolls Royce RB419 units. These are generally similar in concept to the RB410 engines but have been designed specifically to enable large masses of air to be tapped off the compressors. The offtake air is passed through ducts located within the engine mounting pylons and wing before being expelled through a long spanwise nozzle formed by the separated upper and lower surfaces of the trailing edge flap system.

The augmenter wing has one major advantage relative to the externally blown flap system. As the four engines can feed into a single spanwise duct system the effect of a single powerplant failure is much less severe. It may also be possible to produce a quieter aircraft as it is conceivably possible to apply sound treatment to the augmenter system and thereby reduce scrubbing noise which may be a serious difficulty with the externally blown arrangement. Against these advantages must be placed the demands made upon internal volume by the duct system and the mechanical complexity of the flaps.

3. Design conditions

The aircraft is designed to operate from 2000 ft long runways and have a comfort limited cruise speed of 300 knots equivalent airspeed, or $M = 0.83$ whichever is the least. Taken together the runway length and cruise speed limitations are the dominant influences in the design.

In order to achieve a still air landing on a 2000 ft long runway with the usual margins the aircraft is designed to descend along a 7.5 degree glideslope with a 0.25g incomplete flare and a final touchdown vertical velocity of 4 ft/sec. The mean longitudinal deceleration after touchdown is limited to 0.33g by passenger comfort considerations. The requirement to operate from single runway STOL ports implies a need to be able to cope with 20 degrees of sideslip if an acceptably high reliability of operation is to be achieved.

The aircraft is designed to meet the B.C.A.R. requirements in as far as they are applicable to this type of design. Design life for the airframe is 40,000 hours with an average flight duration of 40 minutes. A cabin differential pressure of 8 lb/sq in enables the cabin altitude to be maintained at 6000 ft for all normal operations but during a long range fast cruise it may reach 8000 ft.

The steep approach and difficult flare set the vertical descent velocity at 18 ft/sec, and the cross wind landing implies a need for the main undercarriage wheels to be steered up to 20 degrees in either direction. The main undercarriage can absorb the vertical energy in a landing when the aircraft fails to carry out the flare manoeuvre.

4. Description of aircraft

The configuration of the A71 design is shown in Figure 1. This and the following description applies primarily to the externally blown flap version but the augments wing alternative is similar in most respects.

The design take off weight is 115,000 lbs and the installed static thrust/weight ratio in this condition is approximately 0.5. Design landing weight is 100,000 lbs. Details of the weight of individual components are given in Table 1 and geometric data for the aircraft in Appendix A. Inertia characteristics appear in Table 2.

Sweepback is used in the wing configuration for the following reasons:-

- a) The spanwise flow outwards towards the tips assists in increasing the effectiveness of the thrust deflection system.
- b) The lower lift curve slope is beneficial in reducing gust sensitivity in the cruise. This is of special importance as it places a lower bound on wing area which is best made as high as possible to reduce the magnitude of the required low speed lift coefficient. The relatively low aspect ratio of 5.9 was chosen for the same reason.

- c) The swept wing enables the long range high speed cruise to be flown at rather more than $M = 0.8$. Thus the aircraft is potentially as fast as existing short range types although it must be accepted that the cruise equivalent airspeed limitation implies flight at approximately 30,000 ft altitude for this to be so.
- d) Passengers are now used to flying in swept wing aircraft and will expect new designs to possess this characteristic.

The high mounting of the wing is inevitable because of the need to provide adequate ground clearance for the relatively large diameter powerplants. The considerable downwash effects from the high lift system require the tailplane to be located well away from the wing plane in the vertical sense and the only possible position for it is at the top of the tail fin. Cross wind landing at low approach speed necessitates flight at unusually high sideslip angles and the extensive dorsal fin has been incorporated in the layout to ensure a high fin stall angle.

The fuselage layout is shown in Figure 2. The passenger accommodation is based on the use of six abreast tourist class seating with a single central aisle. Overall fuselage diameter required for this with the high wing configuration is 12.5 ft. When a seat pitch of 33 inches is employed it is possible to carry 120 tourist class passengers. Access is through a forward side door and a rear ventral door. Baggage holds are incorporated in the layout below the passenger floor and an auxiliary power unit is mounted in the tail cone.

Undercarriage design and layout present serious difficulties. The large design vertical descent velocity implies the need for a very long stroke undercarriage to minimise structural fatigue and passenger discomfort. The large cross wind components at landing suggest the necessity for a wide track. Thus the use of fuselage mounted main undercarriage units is not possible and the A71 employs long, inevitably heavy, wing mounted main undercarriage units. As shown in Figure 3 they retract forwards into wing fairings

which do not interfere with the trailing edge flaps but do interrupt the leading edge devices. Four wheel bogie units capable of being preset at steering angles of up to 20 degrees are used for compactness. The nose undercarriage has normal steering capability and is retracted forwards into the fuselage below the crew compartment.

The use of a variable incidence wing was considered in the initial design phase, but it was found to be impracticable. Apart from introducing difficulties with the wing mounted undercarriage the relative rotation of the fuselage brought the tailplane into an unacceptably high downwash field. In any case calculations on the low speed configuration of the aircraft showed that it was possible to arrange for the fuselage to remain in a substantially horizontal position during the approach and thus variable incidence is not required.

5. Control considerations

During cruising and climbing flight the aircraft is controlled by conventional ailerons, rudder and tailplane/elevator combination. The tailplane incidence is adjustable for trim purposes. Airbrakes are located above the wing trailing edge flap for speed control although with variable pitch fans it is likely that the main use of these will be as spoiler/lift dumpers at low speed.

Control of the aircraft at low speed is complicated by the nature of the high lift system and the severe cross wind requirement. The externally blown flaps give a substantial measure of direct lift control which interacts with speed control. Initial calculations suggested that the conventional controls are of insufficient power to deal with the low speed problem and this aspect of the design is the subject of a special investigation.

6. Aerodynamic characteristics

The estimated aerodynamic characteristics of the aircraft are stated in Appendix B and Figures 4 to 7. Aerofoil section ordinates are quoted in Table 3. A study of the low speed stability characteristics is included in the special control investigation.

The data applicable to low speed flight with the flaps deployed has been derived from an interpretation of the N.A.S.A. wind tunnel work on models of aircraft of similar configuration. (2) to (7)

7. Performance

7.1 Take off.

The take off wing loading is 74 lb/sq ft and the nominal thrust/weight ratio 0.5. Take off procedure is for the leading edge flaps to be deployed and the trailing edge flaps set at 10 degrees plus an additional 10 degrees on the aft segment. The engine thrust deflectors are in the cruise position. During the ground roll the aircraft reaches 1.2 times the flaps out stalling speed at which point the engine thrust deflectors are repositioned and rotation takes place. Initial normal acceleration is 0.25g but forward acceleration is small which explains the necessity for rotation to occur at the take off safety speed. In the event of an engine failure before rotation the aircraft can be brought to rest before the end of the 2000 ft runway. Engine failure after rotation necessitates an unaccelerated climb out. The take off safety speed is about 96 knots, and the lift coefficient at rotation just over 3. Further work has shown the need to increase thrust.

7.2 Cruise

Maximum cruise Mach number is 0.83 at 30,000 ft altitude. This condition is thrust as well as Mach limited and can only be achieved at a relatively low flight weight. The normal cruise Mach number at 30,000 ft is 0.8. As the cruise speed is limited to 300 knots equivalent air speed for comfort reasons the useful Mach number is restricted below 30,000 ft, as is shown in Figure 8. Flight at $M = 0.67$ and 20,000 ft is a more usual cruise condition for short stage length operations. The still air, no reserve, payload-range characteristics for both 20,000 ft and 30,000 ft cruise are shown in Figure 9.

The high installed thrust/weight ratio results in an unusually high value of the maximum continuous engine operating speed, V_{M_0} , at low levels. On this basis the design value of the cruising speed, V_c , is approximately 435 knots equivalent air speed and the corresponding design diving speed,

V_D is 485 knots equivalent air speed. The variation of these with altitude is shown in Figure 8. There is no operational requirement to fly at these high air speeds at low level and it would appear to be reasonable to introduce a performance restriction limiting V_{M_0} to approximately 390 knots equivalent air speed and V_D would be correspondingly reduced to 435 knots equivalent air speed or $M = 0.9$ at higher altitudes.

7.3 Landing

At the maximum landing weight the wing loading is 64 lb/sq ft. The approach speed has to be restricted to 79 knots to achieve a landing from 35 ft altitude in 2000 ft with the normal margins. The corresponding approach lift coefficient is 3.4. This is achieved by deploying the leading edge flaps, using the engine thrust deflectors and setting the trailing edge flaps at the 20 degrees plus 20 degrees position. Use of greater trailing edge flap settings introduces speed control difficulties due to the combination of high effective induced drag and low effective forward thrust. It also implies a fuselage attitude which is nose down relative to the ground during approach and this could introduce nose undercarriage design problems in the event of a late flare out.

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TABLE 1
Component Weights

Component	Weight lb	A.U.W. %
Wing, including fairings	11000	9.6
Fuselage	10400	8.9
Tailplane	2140	1.9
Fin	1800	1.6
Main undercarriage	4600	4.0
Nose undercarriage	800	0.7
Structure	30740	26.7
Propulsion engines, complete pods	16000	13.9
Pylons	400	0.3
Engine controls and systems	900	0.8
Powerplant	17300	15.0
Fuel system	1200	1.0
Power supplies	4000	3.5
Auxiliary power unit	530	0.5
Flying control systems	3000	2.6
Deicing and miscellaneous systems	830	0.7
Air conditioning	1500	1.3
Systems	11060	9.6
Radio and radar	1500	1.3
Instruments and automatic units	600	0.5
Fixed equipment	2100	1.8
Sound proofing	800	0.7
Flight crew furnishing	400	0.3
Cabin furnishing	1880	1.6
Cabin seats	2500	2.2
Cabin services, etc	1870	1.6
Furnishings	7450	6.5

... continued

TABLE 1 continued

Component	Weight lb	A.U.W. %
Basic operating empty weight	68650	59.7
Passenger service items, supplies	1250	
Crew	1100	
As prepared for service weight	71000	61.7
Passengers, 120 maximum	24000	20.9
Fuel	20000	17.4
All up weight	115000	100
Passengers, 80	16000	13.9
Fuel, maximum	28000	24.4
All up weight	115000	100

TABLE 2
Moments of Inertia

(Relative to As prepared for service centre of gravity position)

GENERAL

Configuration	Moment of Inertia 10^6 lb ft^2		
	Pitch	Roll	Yaw
As prepared for service, 71,000 lb	33.5	24	52.5
Increment due to 120 passengers, 24,000 lb	10.5	1	10.5
Increment due to 20,000 lb fuel	1	12.5	12.0
Increment due to 28,000 lb fuel	1	17.0	16.5

APPROACH CONDITION - 100,000 lbs

	Speed kts	Trimmed Attitude to flight path	Moment of Inertia 10^6 lb ft^2
Roll-Yaw Product	60	12.55°	-4.15
	70	10.5°	-3.20
	80	8.55°	-1.34
	90	6.20°	-0.48
	100	4.80°	+0.88
Pitch			44.4
Roll			28.1
Yaw			66.0

TABLE 3

Aerofoil Section Coordinates

10% Thickness Chord Ratio

	2% Camber		Symmetrical
% Chord	Upper Surface	Lower Surface	Half Depth
Nose radians	0.56		0.56
0.75	1.10	0.96	1.03
1.25	1.59	1.37	1.48
2.5	2.22	1.84	2.03
5.0	2.96	2.32	2.64
7.5	3.59	2.73	3.16
10	4.05	3.01	3.53
15	4.78	3.42	4.10
20	5.30	3.70	4.50
25	5.61	3.81	4.71
30	5.87	3.91	4.89
35	6.02	3.96	4.99
40	6.06	3.92	4.99
45	5.94	3.74	4.84
50	5.66	3.46	4.56
55	5.30	3.20	4.20
60	4.86	2.72	3.79
65	4.33	2.30	3.33
70	3.81	1.87	2.84
75	3.33	1.53	2.43
80	2.70	1.20	1.90
85	2.10	0.76	1.43
90	1.45	0.51	0.98
95	0.82	0.18	0.50
100	0	0	0

Appendix A. Geometry and Weights - Externally blown flap aircraft.

1 Wing

Gross area	1560 sq ft.
Span	96 ft.
Aspect ratio	5.9
Leading edge sweepback	28°
Root chord (centreline nominal)	24.1 ft
Tip chord (nominal)	8.5 ft.
Standard mean chord, \bar{c}	16.3 ft

Aerofoil sections;

Root: 13% thickness at 37.5% \bar{c} ,
2% camber

Tip: 10% thickness at 37.5% \bar{c} ,
2% camber

See Table 3. Linear Spanwise variation.

Wing-body angle (chord datum to fuselage datum)	0°
Anhedral	3°
Location of 0.25 \bar{c} aft of fuselage nose	49.0 ft
Location of chord datum above fuselage datum	5.62 ft
Location of 0.25 \bar{c} aft of nominal centreline leading edge	14.7 ft

2 Ailerons

Type:- Round nose

Aileron chord/wing chord	0.3
Movement	±20°
Inboard end relative to aircraft centreline	37.6 ft
Outboard end relative to aircraft centreline	47.6 ft

3 Trailing edge flaps

Type: Externally blown, double slotted

Total flap chord/wing chord, retracted 0.365

Subsidiary rear flap chord/total flap chord 0.56

Take off flap setting 10° + 10°

Landing flap setting 20° + 20°

Inboard end of flap from aircraft centreline 6.25 ft.

Outboard end of flap from aircraft centreline 37.5 ft.

4 Leading edge flaps, inboard

Type: Variable camber, Kruger.

Flap chord/wing chord	0.15
Take off flap setting	60°
Landing flap setting	60°
Inboard end of flap from aircraft centreline	6.25 ft.
Outboard end of flap from aircraft centreline, approx.	29 ft

5 Leading edge flaps, outboard

Type: Variable camber, Kruger

Flap chord/wing chord	0.30
Take off flap setting	45°
Landing flap setting	45°
Inboard end of flap from aircraft centreline, approx.	31 ft
Outboard end of flap from aircraft centreline, approx.	47 ft

6 Spoilers

Spoiler chord/wing chord	0.10
Maximum movement	30°
Leading edge of spoiler aft of wing leading edge	0.62c
Inboard end of leading edge relative to aircraft centreline	6.25 ft
Outboard end of leading edge relative to aircraft centreline	37.5 ft

7 Tailplane

Gross area	525 sq ft
Span	45.8 ft
Aspect ratio	4.0
Sweepback of leading edge	28°
Root chord (centreline)	14.3 ft
Tip chord (nominal)	8.6 ft
Aerofoil section: 12% thickness at 37.5% c, symmetrical (see Table 3)	
Dihedral	+3° nose up
Movement	-12° nose down

Tailplane continued

Vertical location of tailplane chord datum above fuselage datum	26.0 ft
Distance of tail 0.25c aft of wing 0.25c	49.2 ft
Location of tail 0.25c aft of nominal centreline leading edge	8.8 ft

8 Elevator

Type:- Round nose

Elevator chord/tailplane chord	0.30
Movement	+ 10° down - 30° up

9 Fin

Nominal area above datum root chord, reference	274 sq ft
Height above datum root chord	19.75 ft
Aspect ratio based on above	1.43
Location of datum root chord above fuselage datum	6.25 ft
Datum root chord	17.0 ft
Tip chord(nominal)	10.7 ft
Sweepback of leading edge	35°
Aerofoil section: 13% thickness at 37.5%oc, symmetrical (see Table 3)	
Distance of leading edge intersection with fuselage datum aft of nose	75.0 ft

10 Rudder

Type: Round nose

Rudder chord/fin chord	0.40
Height of rudder root leading edge above fin root chord	0 ft
Height of rudder tip leading edge above fin root chord	18.5 ft
Movement	± 20°

11 Fuselage

Overall length	96.6 ft
Maximum diameter	12.5 ft
Maximum cabin internal width	11.65 ft
Cabin height	6.5 ft
Cabin length, overall	70.3 ft

12 Undercarriage (See Fig.A71-2 for geometry)

Type: Nosewheel

Wheelbase (to centre of main unit bogie) 41.9 ft

Track (to centre of mainwheels 25.1 ft

Main undercarriage units (See Fig.A71-5)

4 wheel bogie arrangement, forward retracting.

Tyres: 34 in dia x 9.25 in width - 16 in rim.

Pressure 150 p.s.i.

Bogie wheelbase 3.35 ft

Bogie track 2.1 ft

Static tyre closure, approx. 0.25 ft

Maximum tyre closure, approx. 0.5 ft

Nominal shock absorber stroke 3.5 ft

Location of leg aft of fuselage nose 53.2 ft

Overall length of retraction fairing 27.5 ft

Depth of fairing, maximum 4.2 ft

Width of fairing 3.35 ft

Nose undercarriage unit

Twin wheels, forward retracting

Tyres: 34 in dia. x 9.25 in width - 16 in rim

Pressure 180 p.s.i.

Wheel track 1.7 ft

Static tyre closure 0.25 ft

Maximum tyre closure 0.5 ft

Nominal shock absorber stroke 3.1 ft

Location of leg aft of fuselage nose 11.9 ft

13 Propulsion engines

Type:	Rolls-Royce RB 410	
Installation:	4 pods below wing	
Bypass ratio, approx.		10
Sea level rated thrust		14,500 lb
Overall length of complete pod		16.0 ft
Overall diameter of pod		6.3 ft
Intake diameter, nominal		4.6 ft
Location of engine centreline below wing chord datum, approx		5.0 ft
Location of pod front face forward of leading edge, approx.		9.0 ft
Location of inboard engine from aircraft centreline		18.5 ft
Location of outboard engine from aircraft centreline		30.0 ft
Sweepback of mounting pylon leading edge approx.		72°
Thickness/chord ratio of mounting pylon		0.12

Auxiliary power unit

Type:	Airesearch GTCP 85C	
Location of A.P.U. above fuselage datum		42 ft
Location of A.P.U. front face aft of fuselage nose, approx.		85.5 ft

14 WEIGHTS, CENTRES OF GRAVITY AND MOMENTS OF INERTIA

Design normal weight at take off		115,000 lb
Maximum landing weight		100,000 lb
Minimum flying weight		72,000 lb
As prepared for service weight		71,000 lb
Maximum payload		24,000 lb
Maximum fuel load		28,000 lb
Weight breakdown - see Table 1		
Centre of Gravity at APS weight relative to 0.25c and fuselage datum,		
Undercarriage retracted;	$\bar{x} = 0.3$ ft aft	
	$\bar{z} = 2.15$ ft above	
Undercarriage extended;	$\bar{x} = 0.97$ ft aft	
	$\bar{z} = 1.57$ ft above	

Appendix B. Aerodynamic Data - Externally blown flap aircraft

1. Inertia characteristics

Allowable centre of gravity range 0.20c̄ to 0.36c̄
 Moments of inertia - see Table 2

2. Lift characteristics

Maximum lift coefficient, basic aerofoil 1.2

Maximum lift coefficient, take off condition,
 flaps 10° + 10° and full thrust 3.2

Maximum lift coefficient, approach condition,
 flaps 20° + 20° and 80% thrust 5.2

Slope of wing body lift curve, a_1 , clean See Fig. 5

Slope of wing body lift curve, flaps
 deployed See Fig. 4

(N.B. Over the range of blowing coefficient, C_μ ,
 considered the effect on lift curve slope is negligible)

Lift coefficient, flaps 10° + 10° $C_L = 0.456 + 0.0914C_\mu + 0.095\alpha$
 (where α is the fuselage angle of attack in degrees)

Lift coefficient, flaps 20° + 20° $C_L = 0.912 + 1.82C_\mu + 0.0955\alpha$

Wing no lift angle, clean, relative
 to wing centreline chord -2.5°

3. Drag characteristics

Drag polars:-

Cruise: $M = 0.80$ and 30,000 ft. $C_D = 0.0266 + 0.081C_L^2$
 $M = 0.67$ and 20,000 ft. $C_D = 0.020 + 0.072C_L^2$

Zero lift drag coefficient increment due to
 undercarriage 0.021

Take off, flaps 10° + 10°, $C_\mu = 0$ $(C_D)_{C_\mu=0} = 0.13 + 0.117C_L^2$
 $= 0.154 + 0.0102\alpha + 0.00107\alpha^2$

Approach, flaps 20° + 20°, $C_\mu = 0$ $(C_D)_{C_\mu=0} = 0.151 + 0.091C_L^2$
 $= 0.227 + 0.01585\alpha + 0.00083\alpha^2$

4. Axial force characteristics

Take off, flaps 10° + 10°

$$C_{FA} = C_\mu (0.81 - 0.0295\alpha + 0.0045C_\mu\alpha - 0.06C_\mu)$$

where C_{FA} is the coefficient of axial force excluding
 the zero below drag coefficient

Approach, flaps $20^\circ + 20^\circ$

$$C_{FA} = C_\mu (0.423 - 0.0466\alpha + 0.00762C_\mu\alpha - 0.0381C_\mu)$$

5. Pitching moment characteristics

Pitching moment coefficient at zero lift,

clean aircraft, C_{M0} -0.07

Location of low speed overall wing-body aero.

centre, clean aircraft, from fuselage nose 48.7 ft

Location of overall wing-body aero. centre,

$M = 0.9$ 49.0 ft

Pitching moment coefficient at zero lift,

$$\text{take off condition flaps } 10^\circ + 10^\circ \quad C_{M0} = -(0.2050 + 0.77C_\mu - 0.07C_\mu^2)$$

Increment at fwd c.g. due to lift

$$\Delta C_M = \left[0.047 + 0.042C_\mu - \frac{0.0314}{(C_\mu + 0.2)} \right] C_L - 0.006C_L^2$$

Increment at aft c.g. due to lift

$$\Delta C_M = \left[0.468 - 0.0115C_\mu - \frac{0.514}{(C_\mu + 1.21)} \right] C_L - 0.0112C_L^2$$

Pitching moment coefficient at zero lift, approach

$$\text{condition flaps at } 20^\circ + 20^\circ \quad C_{M0} = -(0.35 + 1.44C_\mu - 0.11C_\mu^2)$$

Increment at fwd c.g.

$$\Delta C_M = \left[0.06 + 0.0775C_\mu - \frac{0.0185}{(C_\mu + 0.142)} \right] C_L - 0.000832C_L^2(13.48 + C_\mu)$$

Increment at aft c.g.

$$\Delta C_M = \left[0.22 + 0.068C_\mu - \frac{0.0196}{(C_\mu + 0.153)} \right] C_L - 0.00396C_L^2(4.67 - C_\mu)$$

6. Control and stabiliser characteristics, basic surfaces
(per radian)

Location of mean tailplane aero. centre aft

of fuselage nose, cruise 98.6 ft

Location of mean fin aero. centre aft of

fuselage nose, cruise 89.0 ft

Slope of tailplane lift curve, a_{1T} see Fig. 5

Ratio of elevator lift curve slope, a_{2T}/a_{1T} 0.68

Slope of elevator hinge moment curve due to

tailplane incidence, b_{1T} -0.26

Slope of elevator hinge moment curve due to

elevator angle, b_{2T} -0.59

Slope of fin lift curve, a_{1F} (net area and a_{1BT} (including body and tail effect)	See Fig. 5
Ratio of rudder lift curve slope, a_{2F}/a_{1F}	0.83
Slope of rudder hinge moment curve due to fin incidence, b_{1F}	-0.13
Slope of rudder hinge moment curve due to rudder angle, b_{2F}	-0.43
Rolling moment coefficient due to aileron, cruise, l_{ξ}	-(0.045+0.1M)
approach	$-0.125+0.000167\alpha^2-0.0174C_{\mu}$
Slope of aileron hinge moment due to wing incidence, b_1 , $C_{\mu}=0$	-0.31
Slope of aileron hinge moment due to aileron angle b_2 , $C_{\mu}=0$	-0.63
Rolling moment coefficient due to rudder, l_{ζ} , approach	$0.0625-0.00261\alpha$
Yawing moment coefficient due to aileron, n_{ξ} , approach	$0.016+0.013C_{\mu}$
Yawing moment coefficient due to rudder, n_{ζ} , approach	-0.152
Side force coefficient due to aileron, y_{ξ} , approach	0
Downwash at tailplane, cruise	See Fig. 6
approach	See Fig. 7

7. Lateral stability derivatives (per radian)

Rolling moment derivatives due to:-

Roll, l_p , cruise, $0.6 < M < 0.83$ $-(0.27+0.09M)$

approach $-0.00088(481+4.09\alpha-\alpha^2)-0.0025$

$\{C_{\mu}(50.3-5\alpha+\alpha^2)-0.34C_{\mu}^2(35.7-5\alpha+\alpha^2)\}$

Sideslip, l_v , cruise $-[0.01+0.14C_L+a_{1BT}(0.023-0.035C_L)]$

approach $-0.14-0.009\alpha-C_{\mu}[0.071-0.0073\alpha+0.00145C_{\mu}\alpha-0.01C_{\mu}]$

Yaw, l_r , cruise $0.21C_L+a_{1BT}(0.02-0.03C_L)$

approach $0.26+0.013\alpha$

Yawing moment derivatives due to:-

Roll, n_p , approach $-0.135-0.0025\alpha-C_{\mu}(0.00235\alpha+0.0365-0.0121C_{\mu})$

Sideslip, n_v , cruise $0.073a_{1BT}-0.07$

approach $0.166+0.00258\alpha+0.117C_{\mu}-0.025C_{\mu}^2$

Yaw, n_r , cruise $-(0.07+0.18C_L^2+0.061a_{1BT})$
 approach -0.188

Sidelforce derivatives due to:-

Roll, y_p , approach $-0.035+0.317C_\mu-0.171C_\mu^2$
 Sideslip, y_v , cruise $-(0.15+0.176a_{1BT})$
 approach $-0.24+0.00052\alpha-0.000478\alpha^2$
 $-0.078C_\mu+0.00825C_\mu^2$
 Yaw, y_r , approach $0.035-0.0025\alpha$

Tailplane rolling moment coefficient due
 to sideslip, K_β , cruise 0.15

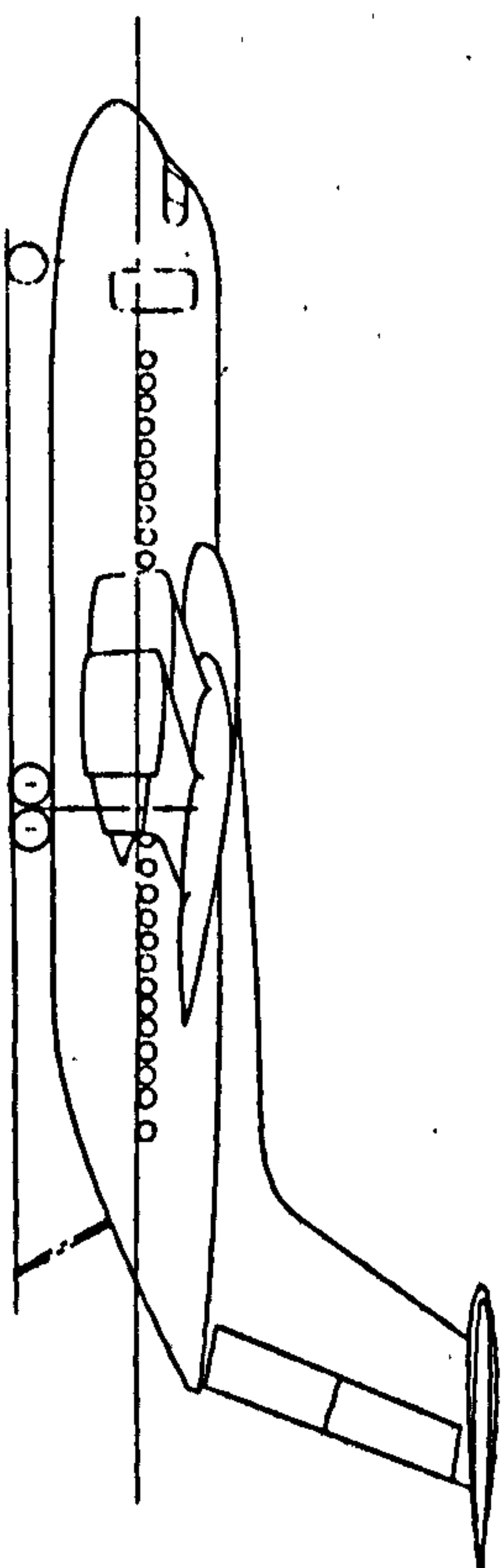
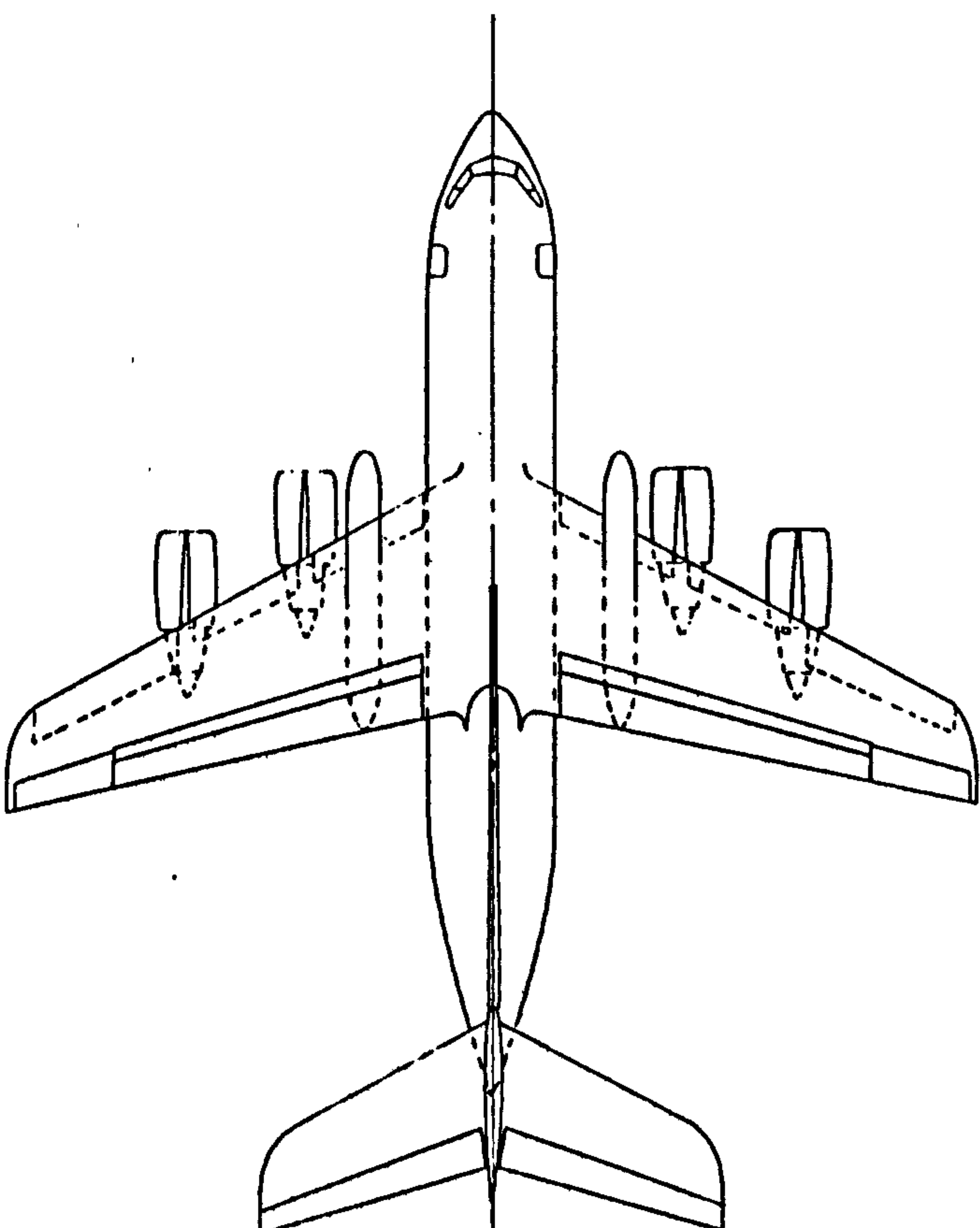
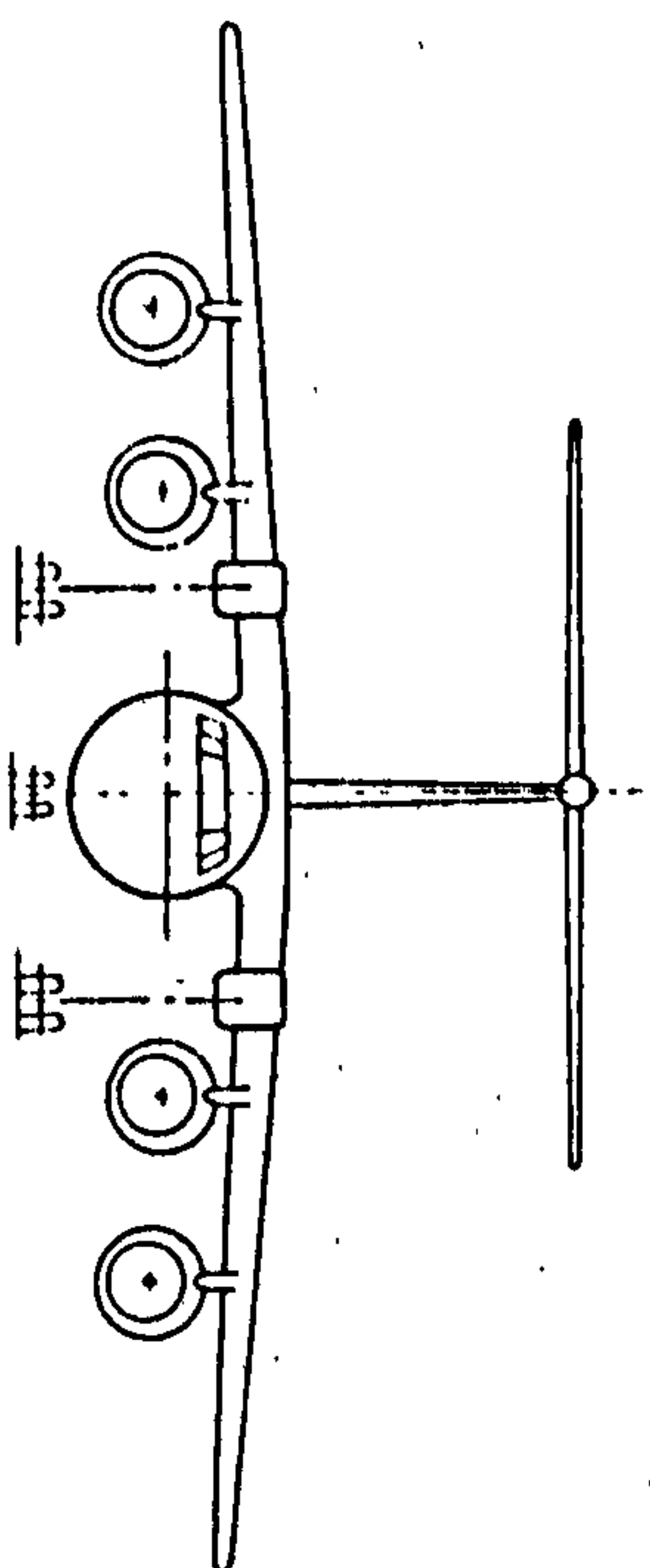


FIGURE 1. GENERAL ARRANGEMENT OF A71 AIRCRAFT

ALL DIMENSIONS IN INCHES

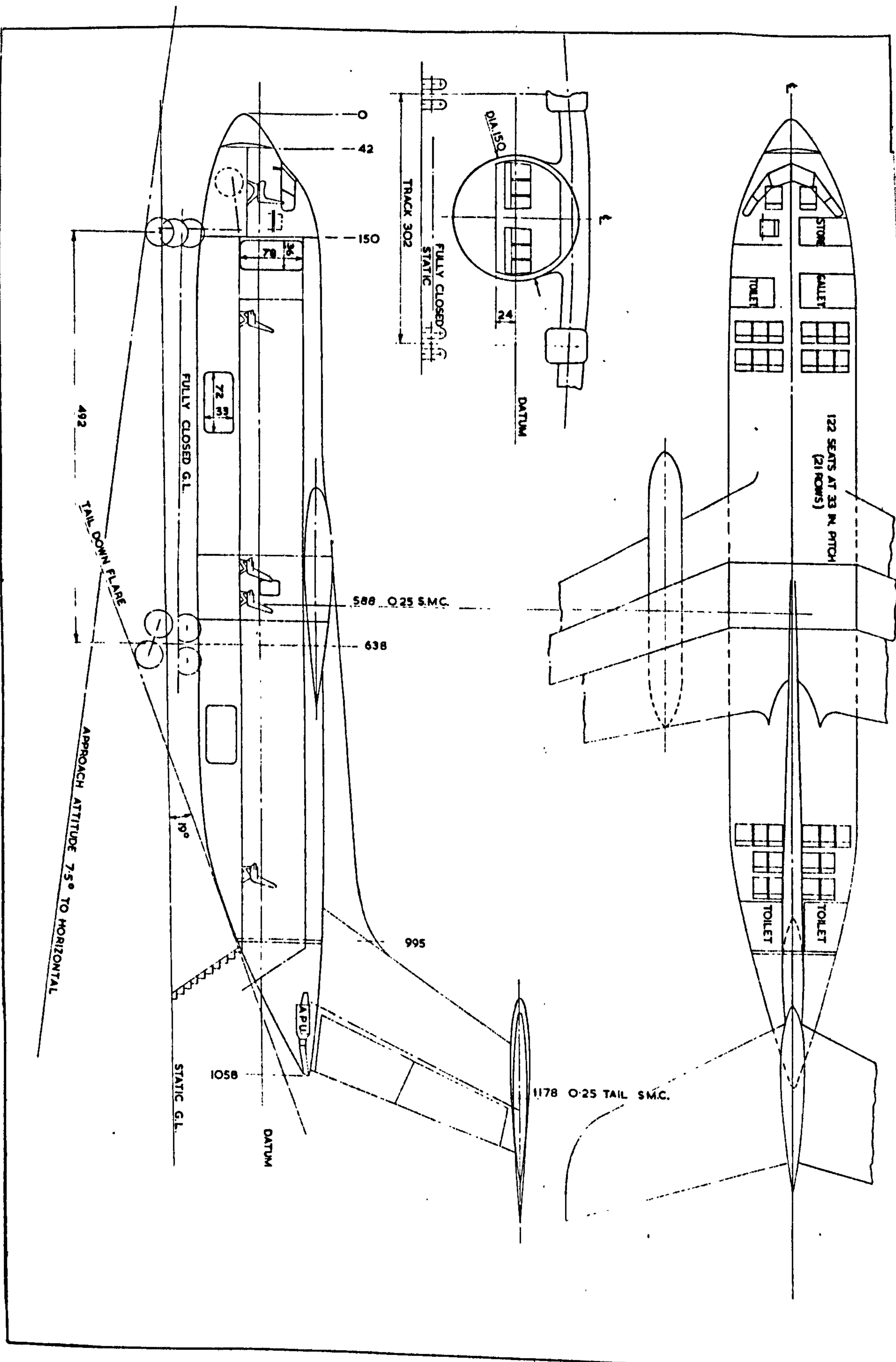


FIGURE 2. FUSELAGE LAYOUT

ALL DIMENSIONS IN INCHES

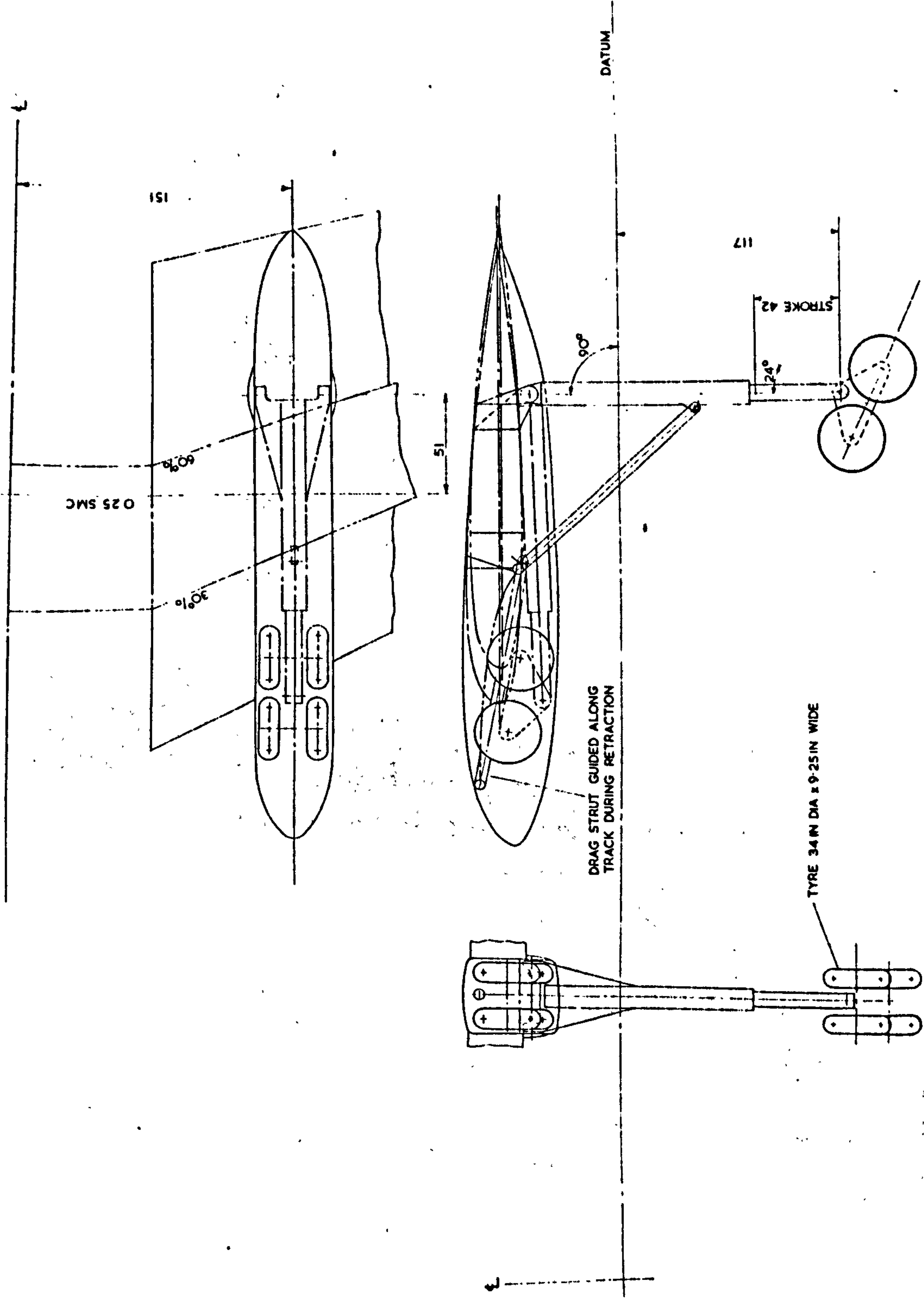


FIGURE 3. UNDERCARRIAGE LAYOUT

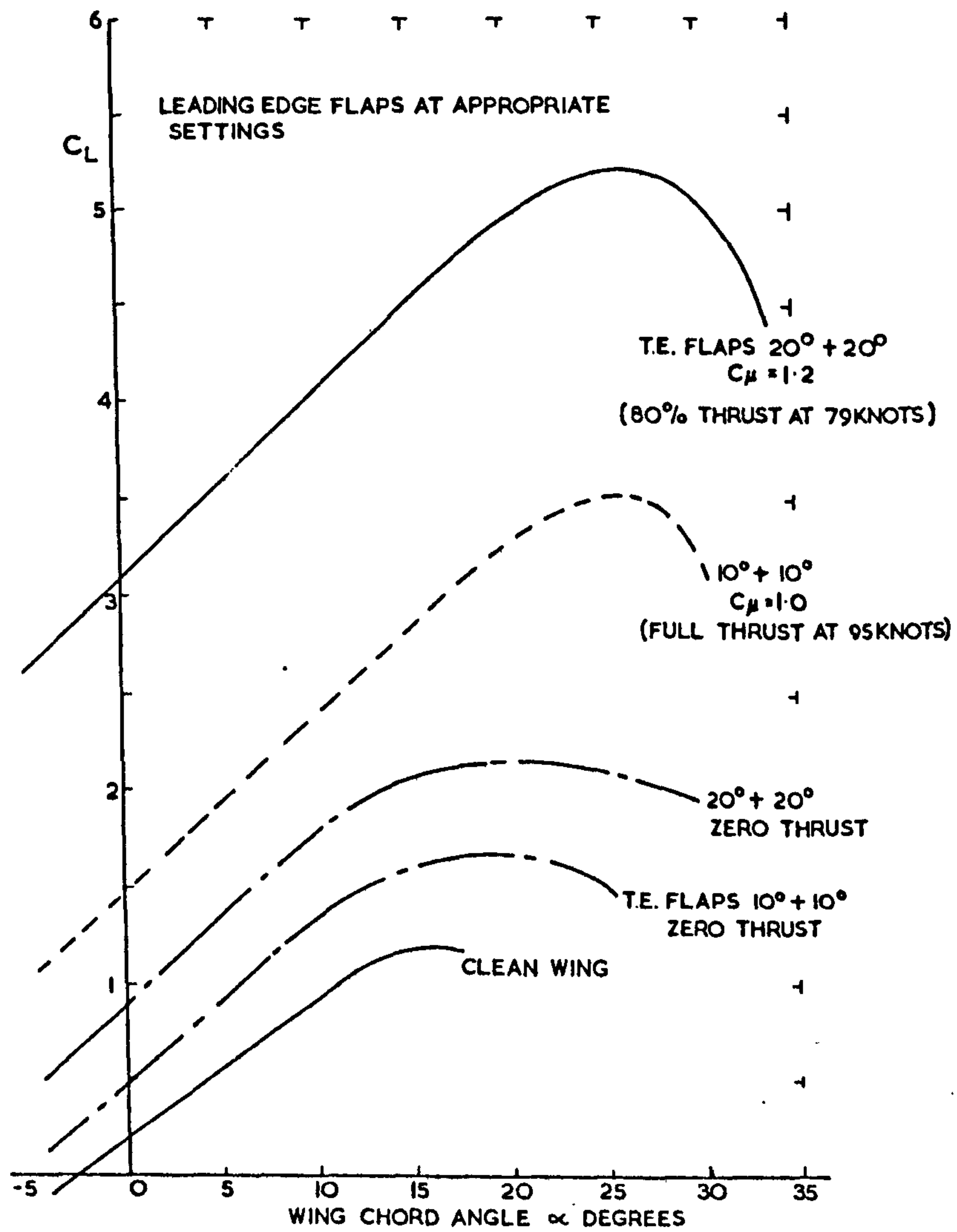


FIGURE 4. LOW SPEED LIFT CHARACTERISTIC

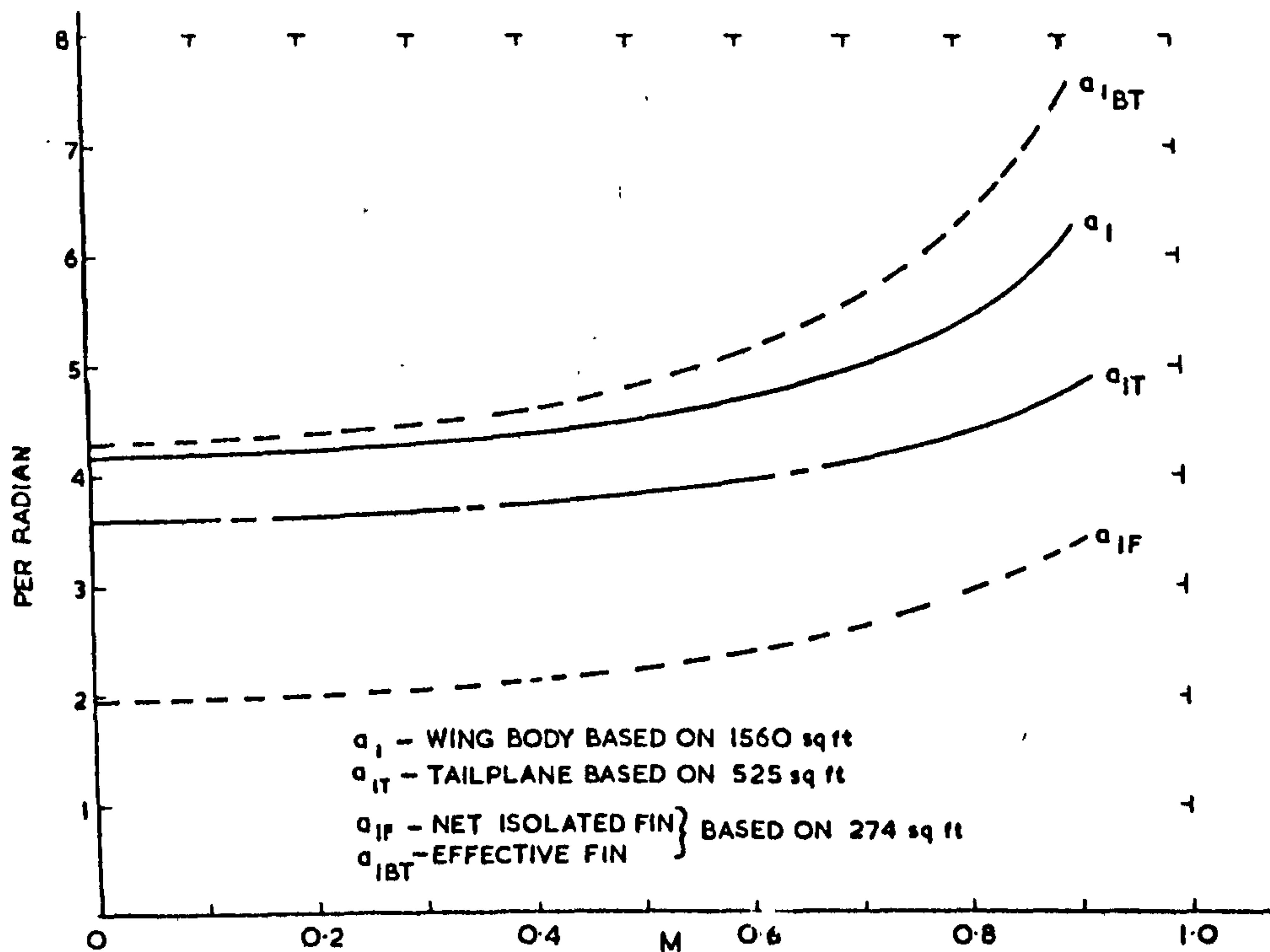


FIGURE 5. CRUISE LIFT CURVE SLOPES

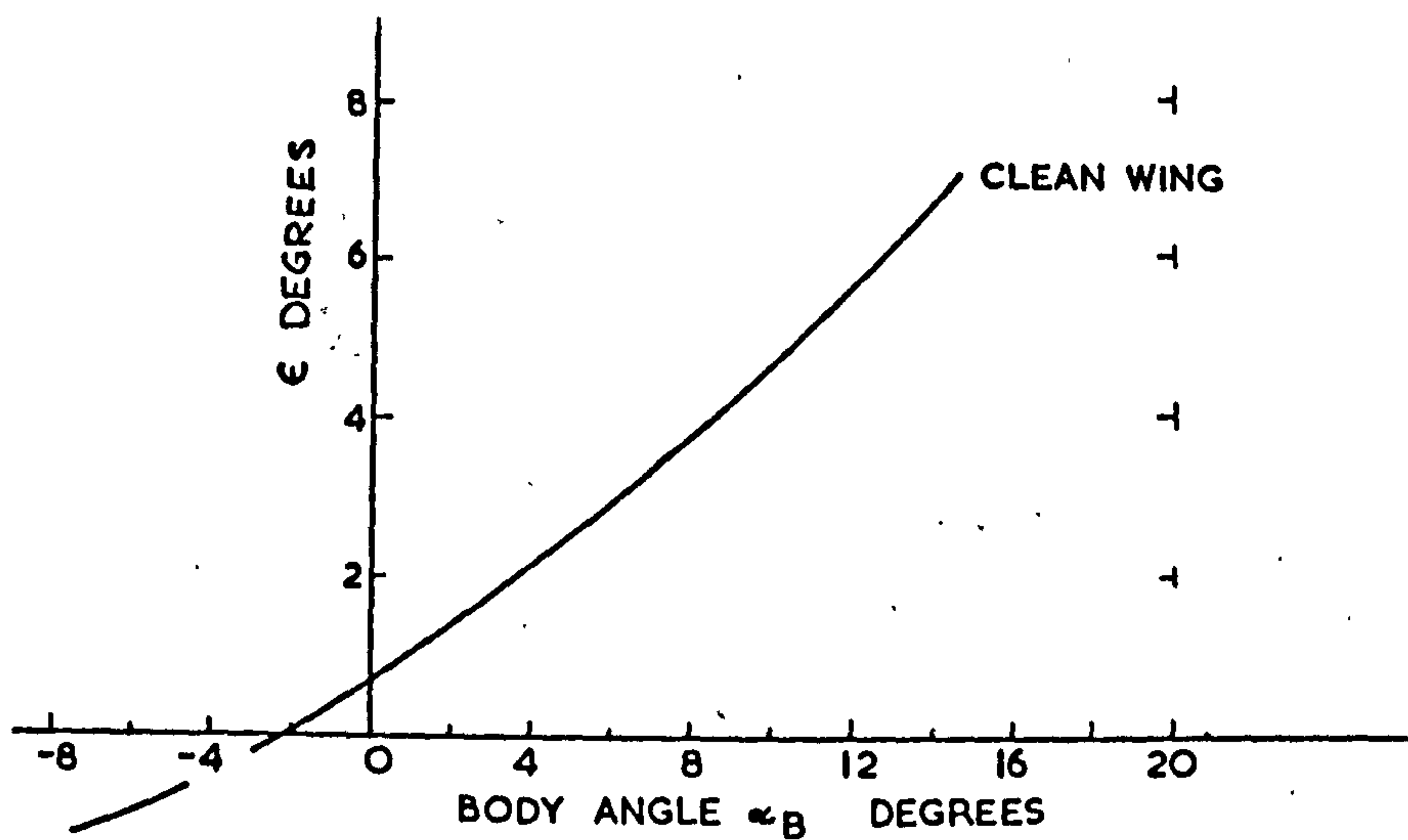


FIGURE 6. DOWNWASH AT TAILPLANE—CRUISE

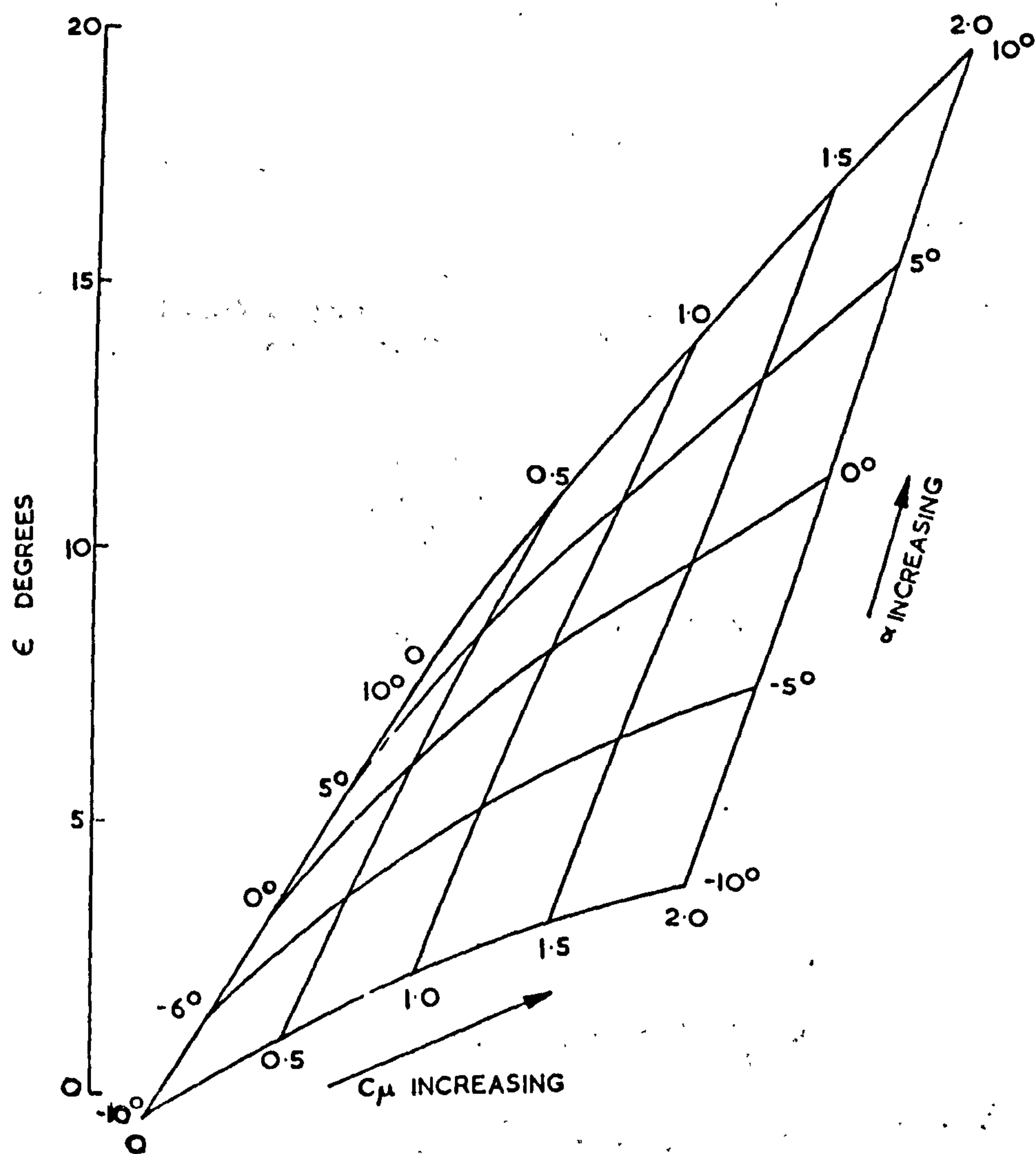


FIGURE 7. DOWNWASH AT TAIL—LOW SPEED

DAMAGED WHEN RECEIVED. REMPLOY

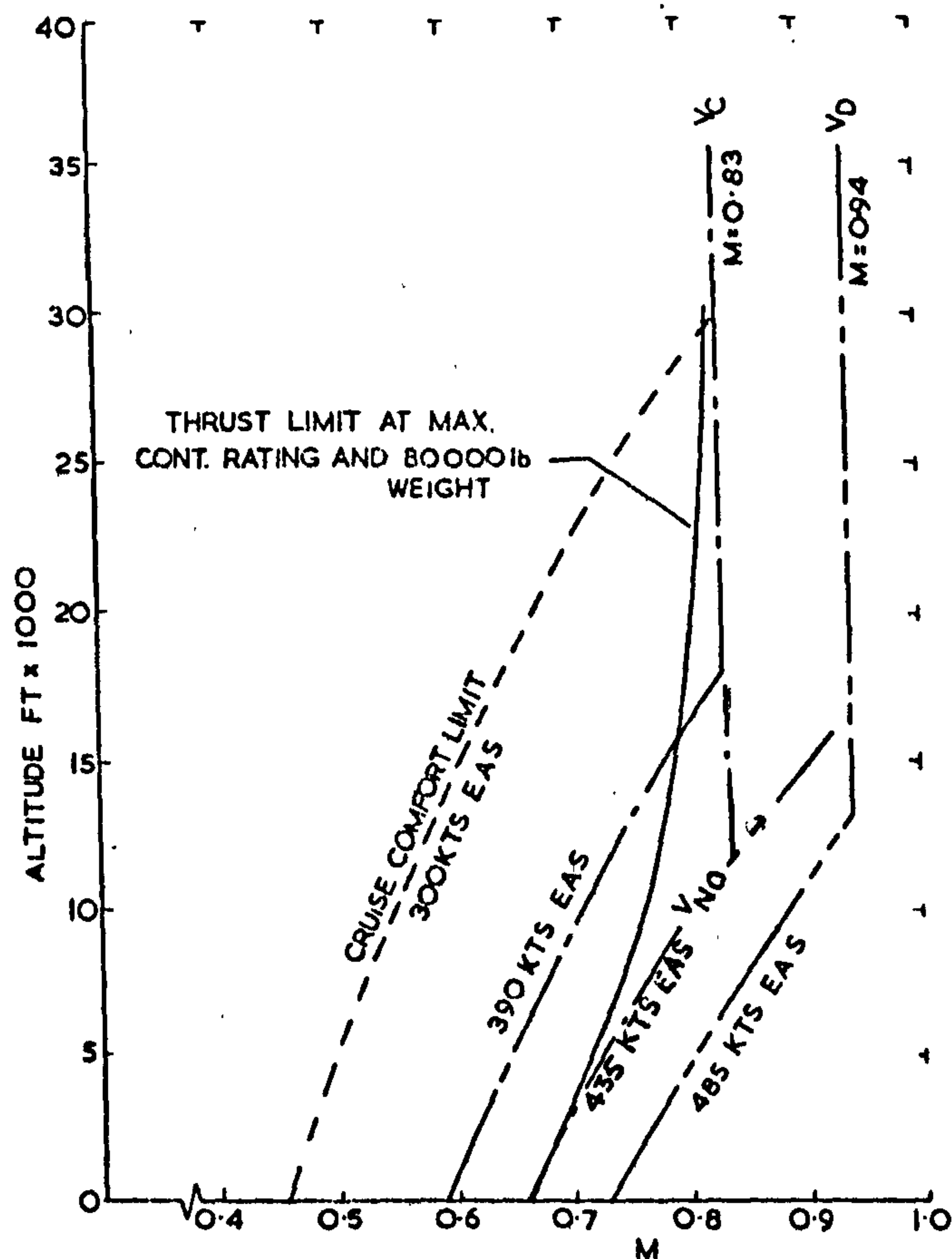


FIGURE 8 LEVEL SPEED PERFORMANCE AND LIMITATIONS

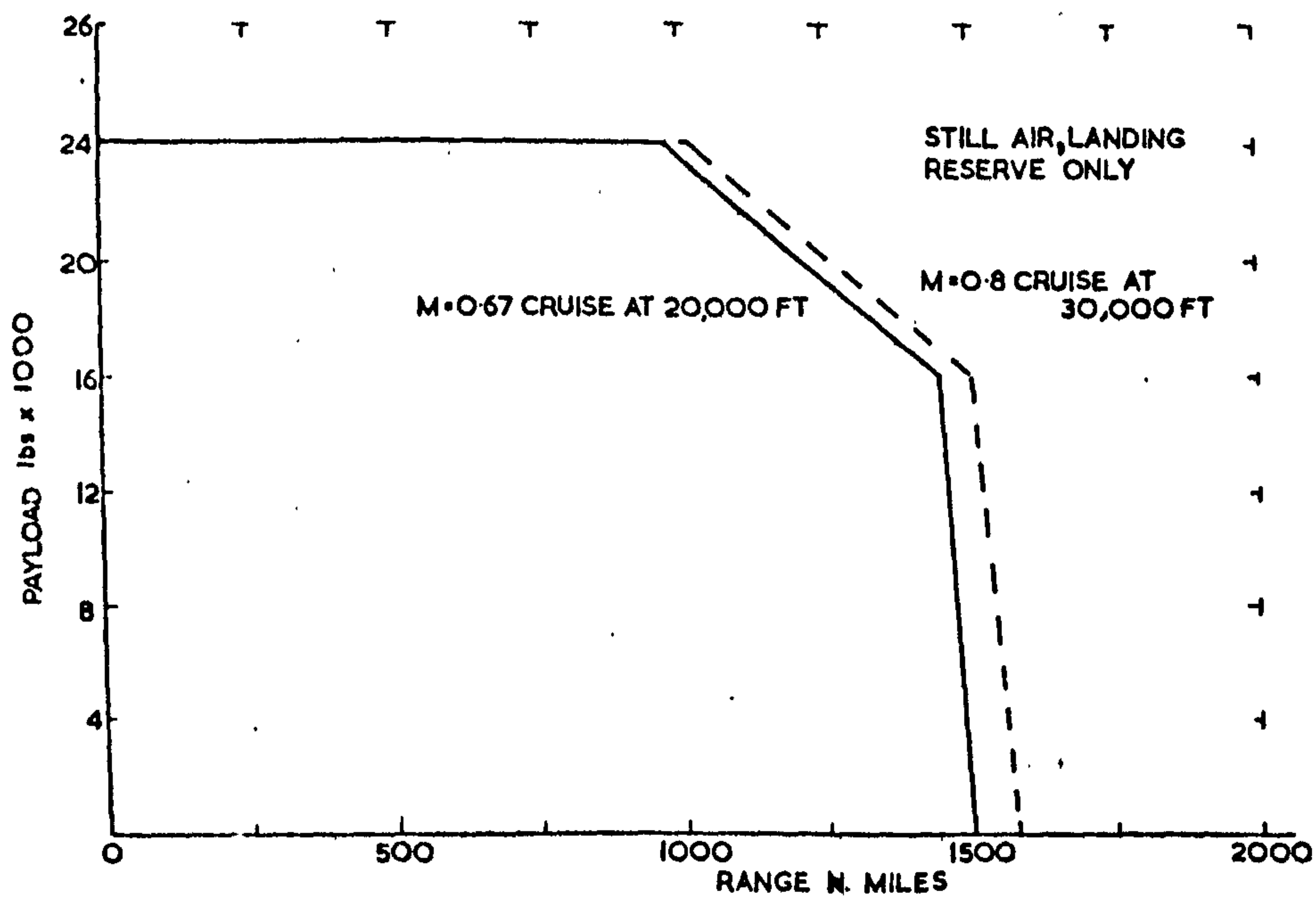


FIGURE 9 PAYLOAD-RANGE PERFORMANCE